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PRELIMINARY CONTINGENCY PROCEDURES FOR THE TRANSLUNAR COAST

MISSION PLANNING AND ANALYSIS DIVISION

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

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PROJECT APOLLO

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By Bobbie D. Weber, Flight Analysis Branch and
Jerry D. Fuller, Contingency Operations Section, TRW Systems Group

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PRELIMINARY CONTINGENCY PROCEDURES FOR THE TRANSLUNAR COAST

By Bobbie D. Weber, Flight Analysis Branch, and
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INTRODUCTION AND SUMMARY

This paper presents detailed procedures to be used in the event of contingencies identified during or after the transposition and docking (T and D) maneuver. This does not include contingencies identified during any malfunction checks made immediately following translunar injection (TLI) cutoff. Procedures for contingencies arising during the TLI burn or the malfunction checks are documented in reference 1. The general ground rules and suggested contingency procedures described below are substantiated by discussion of (1) abort targeting and abort trajectory analysis, (2) ground and onboard return-to-earth programs, and (3) spacecraft systems and equipment capabilities.

The term contingency will be used to specify all those possible problems requiring a spacecraft abort at any time after the block-data abort point at TLI cutoff plus 90 minutes. Conditions resulting in alternate missions are discussed in reference 2 and are omitted here. Reference 3 indicates possible alternate lunar missions that can result from various dispersed TLI trajectories.

The procedures presented in this document are of a preliminary nature. Final procedures will be included in an Apollo Mission contingency procedures document to be published by the Contingency Analysis Section, Flight Analysis Branch.

2. NOMENCLATURE

AAWG	Apollo abort working group
AOT	Alignment optical telescope
AST	Abort scan table
BMAG	Body-mounted attitude gyro
c	Steering constant
CLA	Contingency landing area
CMC	Command module computer
COAS	Crew optical alignment sight
CSM	Command service module
DAP	Digital autopilot
DPS	Descent propulsion system
EMS	Entry monitoring system
FCUA	Fuel-critical unspecified-area
FDAI	Flight director attitude indicator
fps	feet per second
FTP	Full throttle point
g.e.t.	Ground elapsed time
IMU	Inertial measurement unit
ISS	Inertial subsystem
LGC	LM guidance computer
LLM	Lunar landing mission
LM	Lunar module
LOI	Lunar orbit insertion

MCC	Midcourse correction
MGA	Middle gimbal angle
MSFN	Manned spaceflight network
MSI	Moon's sphere of influence
PGNCS	Primary guidance, navigation and control subsystem
PIPA	Pulsed-integrating pendulous accelerometer
PLA	Planned landing area
RCS	Reaction control subsystem
RTCC	Real-time computer complex
RTEAP	Return-to-earth abort processor
RTED	Return-to-earth digitals
SCS	Stabilization and control subsystem
SCT	Scanning telescope
SHe	Supercritical helium
SM	Service module (when used as a subscript: Stable member)
SPS	Service propulsion subsystem
SXT	Sextant
T and D	Transposition and docking
T_{ar}	Time from abort to entry
TCUA	Time-critical unspecified-area
t_d	Delay time from some epoch, e.g. TLI cutoff
TEI	Transearth injection
TFT	Total flight time
t_{go}	Time to go (before thrust cutoff)

t_{ig}	Time of ignition
TLC	Translunar coast
TLI	Translunar injection
TVC	Thrust vector control
V_{EI}	Inertial velocity magnitude
$\vec{V}_g$	Velocity to be gained vector for Lambert aim point guidanc
$\vec{V}_r$	Velocity required vector
$\vec{\Delta V}$	Velocity change vector
ΔV or ΔV_c	Change in velocity magnitude
γ_{EI}	Inertial flight path angle, measured from the local horizontal

3. GENERAL INFORMATION

3.1 Nominal Free-Return Circumlunar Trajectory

The sequence of events for a nominal free-return circumlunar trajectory can be described in terms of a typical lunar landing mission (LLM). Table I gives the ground elapsed time (g.e.t.) for events during the free-return trajectory following a 1 February 1968 launch. The times in table I are taken from reference 4 and correspond to a Pacific injection on the second revolution in earth parking orbit. In this particular example, translunar injection occurs at latitude 0.7 degree S, longitude 179.8 degrees E, and places the spacecraft on an ellipse inclined 54 degrees above the earth-moon plane. Figures 3.1-1 and 3.1-2 show time histories of several trajectory parameters during translunar coast (TLC).

From table I, the flight time from free-return pericynthion to entry is about 72.5 hours, compared with a nominal 106.7 hours from transearth injection (TEI) to entry on the complete LLM. Total flight time (TFT) for the free-return circumlunar trajectory, measured from TLI to entry interface, has a range from approximately 126 hours to 148 hours over an entire yearly launch window. This variation can be inferred from the data of reference 5 and is almost entirely due to variation in the earth-moon distance. A launch azimuth range of 72 degrees through 108 degrees accounts for less than 2 hours variation in this TFT.

Figure 3.1-3 is a schematic diagram of the free-return circumlunar trajectory, as viewed in the rotating earth-moon system. The approximate longitudinal position of Houston at the time of translunar injection is shown on the figure, as well as time ticks indicating the approximate positions of the spacecraft at 24-hour intervals following injection.

3.2 Abort Targeting

The targeting scheme for an abort maneuver must, of prime importance, place the spacecraft on a trajectory which will enter the earth's atmosphere with a flight path angle lying within the entry corridor limits. A secondary objective is the coordination of the spacecraft landing point with available recovery forces.

Targeting parameters may be classified as direct and indirect, where direct parameters are defined to be orbital elements associated with the desired postabort orbit. Components of a desired target vector, e.g. the unit vector in the desired plane for lunar orbit insertion (LOI) circularization guidance, are direct targeting parameters. In general Lambert targeting, the direct parameters are a desired target vector and the time of arrival at that vector. To determine a unique postabort orbit, other, indirect targeting parameters are required. The time of ignition and the cross-product steering constant c are examples of these.

TABLE I. TIME SEQUENCE FOR A FREE-RETURN CIRCUMLUNAR TRAJECTORY - LAUNCH ON 1 FEBRUARY 1968

Event	Ground Elapsed Time hr:min:sec	Approximate Time Difference hr:min
Launch	00:00:00	
		00:11
Earth Parking Orbit Insertion	00:10:58.4	
		02:29
TLI Burn Initiation (Pacific Injection)	02:39:54	
		00:5.5
TLI Cutoff	02:45:23.4	
		07:00
Translunar Midcourse Correction (TLMC) No. 1	09:45:23	
		17:00
TLMC No. 2	26:45:23	
		26:33
TLMC No. 3	53:18:11	
		16:00
TLMC No. 4	69:18:11	
		8:04
Free-Return Pericynthion	77:21:44.7	
		72:27.5
Entry Interface	149:49:14.8	

For a guided burn the vector velocity to be gained at any instant ($\vec{V}_g$) is found by differencing the required velocity at that instant, $\vec{V}_r$, and the present spacecraft velocity, $\vec{V}$. The required velocity vector ($\vec{V}_r$) is a function of the particular set of direct targeting parameters being used. The ultimate objective is to apply the thrust acceleration so as to zero the velocity to be gained ($\vec{V}_g$). In Lambert targeting the direction of $\vec{V}_g$ is allowed to change during the burn, under the control of the steering constant c. Typical values of c are 0 to 1. An operationally simpler guidance scheme is one in which $\vec{V}_g$ is calculated once and the abort thrust vector is fixed in this inertial direction. When the magnitude of $\vec{V}_g$ reaches zero, thrust is terminated. This latter method is called external ΔV maneuver guidance.

Whenever ground/spacecraft communications are available, abort targeting parameters will be uplinked from the ground to the CMC. In the case of Lambert targeting, these parameters will consist of:

(1) Time of abort burn ignition, t_{ig}

(2) Steering constant, c

(3) Components of the target vector

(4) Time from t_{ig} until the target vector is reached.

For external ΔV maneuvers, the uplinked parameters are the time of ignition (t_{ig}) and the components of the required ΔV expressed in local vertical coordinates.

More detailed information on targeting schemes is contained in reference 14, from which part of the foregoing discussion was taken.

Spacecraft aborts from the TLC, performed at 5 hours or more after TLI cutoff, result in inertial entry velocities within the approximate range of 34,500 to 36,300 feet per second. This range of inertial velocities corresponds to the shaded region of figure 3.2-1, which is a plot of inertial flight path angle (γ_{EI}) versus inertial velocity (V_{EI}) at the entry interface (400-000 ft altitude). Figure 3.2-1 shows the 10g full-lift undershoot boundary, the zero-lift overshoot boundary, and two entry target lines. Both the manned spaceflight network (MSFN) target line and the contingency target line are used by the ground-based return-to-earth abort processor (RTEAP), while the onboard return-to-earth program (COLLOSSUS P37) contains only the contingency target line. The onboard target line is defined to be at the geometric center of the most restrictive entry corridor (reference 6), which is bounded by the zero-lift overshoot boundary and the 12g undershoot boundary.

In addition to those built-in target lines, there exists, in each of these programs, the capability to accept a manually-entered value for γ_{EI}. This γ_{EI} override capability might be used, for example, to increase the probability of capture following a time-critical high-ΔV abort maneuver performed without a current navigation update.

3.3 Abort Trajectory Analysis

3.3.1 Unspecified-area abort- This is the least restrictive abort solution available for any given situation, since there are no landing constraints whatever. The solution is constrained to be inplane unless a plane change is required to satisfy the constraint on maximum return inclination (typically 40 degrees). If the preabort plane of motion does not require a plane change, the solution is generated using either minimum time or minimun fuel. The minimum-time solution uses all the available ΔV to obtain the shortest possible postabort flight time, within the maximum entry speed and minimum return time restrictions imposed. If either of these constraints is encountered, a smaller ΔV is used, viz., that ΔV corresponding to either one or both of these limiting values. The minimum-fuel solution employs the least ΔV possible to obtain a return trajectory within the constraints imposed on maximum and minimun return time and maximum entry speed.

In practice, all ground-computed abort solutions are constrained to land on water, to maximize the probability of a safe landing and quick recovery. This constraint may be imposed by approximating the earth's land mass boundaries by straight line segments and requiring that at least part of the available landing footprint lies outside these boundaries. Figure 3.3-1 is a typical representation of land masses taken from section 4 of reference 15, which also contains a complete discussion of the methods of abort solutions.

Representative abort trajectories from TLC are schematically illustrated in figure 3.3-2. The trajectories correspond roughly to an abort ΔV of 1900 feet per second in order to simulate maneuvers limited to the LM descent propulsion system (DPS) ΔV capability. At a fixed abort ΔV of this magnitude, the total flight time (TFT) from TLI cutoff to entry increases rapidly with preabort delay time (t_d). This increase is shown in figure 3.3-3, taken from reference 6. In figure 3.3-4 postabort flight time (T_{ar}) is shown versus t_d over the first 20 hours of TLC, including the 1900 fps abort case. With this ΔV, and t_d between 18 and 25 hours following TLI cutoff, the abort TFT reaches a value equal to the free-return circumlunar TFT. This 7-hour range of t_d is due to the earth-moon distance variation mentioned in the previous section. For delay times longer than 18-25 hours, direct-return aborts using 1900 fps result in longer TFT's than those performed post-pericynthion on the circumlunar trajectory.

3.3.2 Landing Site Control- Though not the least restrictive abort solutions, aborts restricted only to water landings are in general too demanding from the point of view of recovery. Increased landing site control is exercised in one of two ways: by constraining the center of the landing footprint to a specific longitude, i.e., to a contingency landing area (CLA); or by constraining the center of the landing footprint to be within a given radius about a specific longitude and latitude (planned landing area, PLA). A PLA solution with a zero allowable miss is the most restrictive abort solution possible. Restrictions on maximum and minimum return times, maximum entry speed, and maximum entry inclination apply also to CLA and PLA solutions.

Landing site analysis for TLC abort maneuvers is reported in detail in reference 7, and the following discussion is partially abstracted from that document. The TLC is a portion of an ellipse having a very large period and a high apogee altitude. This ellipse is relatively fixed in inertial space, with its apogee direction tilted 6 to 11 degrees from the earth-moon line at the time of free-return pericynthion. As a result, the geocentric inertial entry point following abort maneuvers from TLC is relatively fixed, i.e., insensitive to abort delay time and, to a lesser extent, to abort ΔV. An abort to a specific contingency landing area (CLA) therefore requires a trajectory which reaches this fixed inertial entry point simultaneously with the longitude of the CLA. A wide choice of solutions results from the tradeoff between abort ΔV and TFT as the abort is delayed along the TLC. Figures 3.3-5 through 3.3-9 are taken from reference 6 and show the relationships among abort ΔV, TFT and t_d for four different CLA lines (figure 3.3-10). These figures also show the 24-hour breaks in TFT for a specific CLA, due to the fixed inertial entry position of the postabort trajectories. Minimum-time returns to the four CLA lines are summarized in figure 3.3-9. Note that none of the minimum-time solutions are available for an abort ΔV of 1900 fps.

Aborts to a PLA differ from CLA aborts primarily in the required plane change to reach a specific latitude. The tradeoff involved between abort ΔV and TFT is similar to that for CLA maneuvers, but the ΔV requirements are higher. Delta velocity difference between PLA and CLA aborts to a common longitude is a sensitive function of the latitude of the PLA, being zero for a PLA lying in the preabort plane. For example, a CLA abort at TLI cutoff plus 5 hours uses a ΔV of 1757 fps to reach a landing point at longitude 65 degrees E. The associated total flight time is 78.8 hours and the landing point latitude is 4.86 degrees S. A typical corresponding PLA differs only by 7.64 degrees in latitude, being located at latitude 12.5 degrees S. To reach this PLA requires a ΔV of 1897 fps and a TFT of 74.0 hours.

The tradeoff between delay time and total flight time can be used to reach any desired CLA with a fixed ΔV expenditure. The resulting spread in TFT is approximately 16 to 19 hours. Table II gives TFTs to the four CLA lines defined in reference 7 for the indicated ΔV values and intervals of t_d. Similar information for three PLAs is shown in table III.

Midcourse ΔV requirements following perturbed abort maneuvers are discussed in reference 8. One-degree thrust vector misalignments were simulated in unspecified-area and PLA-type aborts from the TLC. The minimum required ΔV for a postabort midcourse correction (MCC) is, in general, a parabolic appearing function of postabort orbital position, with a minimum at or near postabort apogee. This behavior alters as the spacecraft approaches the earth, and the MCC ΔV increases sharply. Following perturbed aborts from TLC, the original abort landing point can in most cases be maintained to within 50 n.mi. with a MCC ΔV budget of 150 fps, applied in a minimum-fuel unspecified-area mode.

TABLE II. TOTAL FLIGHT TIMES FOR CLA ABORTS

Abort ΔV (fps)	Minimum Delay from Cutoff (hrs)	TFT (hrs) CLA A 20°W	CLA B 65°E	CLA C 176°W	CLA D 100°W	TFT Spread (hrs)
1900	5	85	79	95	90	16
	10	109	103	119	114	16
	15	133	127	119	138	19
	20	133	151	143	138	18
4000	30	109	103	119	114	16
	40	133	127	119	138	19

TABLE III. TOTAL FLIGHT TIMES FOR PLA ABORTS

Abort ΔV (fps)	Minimum Delay from Cutoff (hrs)	TFT (hrs) PLA 1 176°W,11°S	PLA 2 20°W,15°S	PLA 3 65°E,15°S	TFT Spread (hrs)
1900	5	95	109	103	14
	10	119	109	127	18
	15	119	133	127	14
	20	143	133	151	18
4000	30	119	107	103	16
	40	119	133	127	14

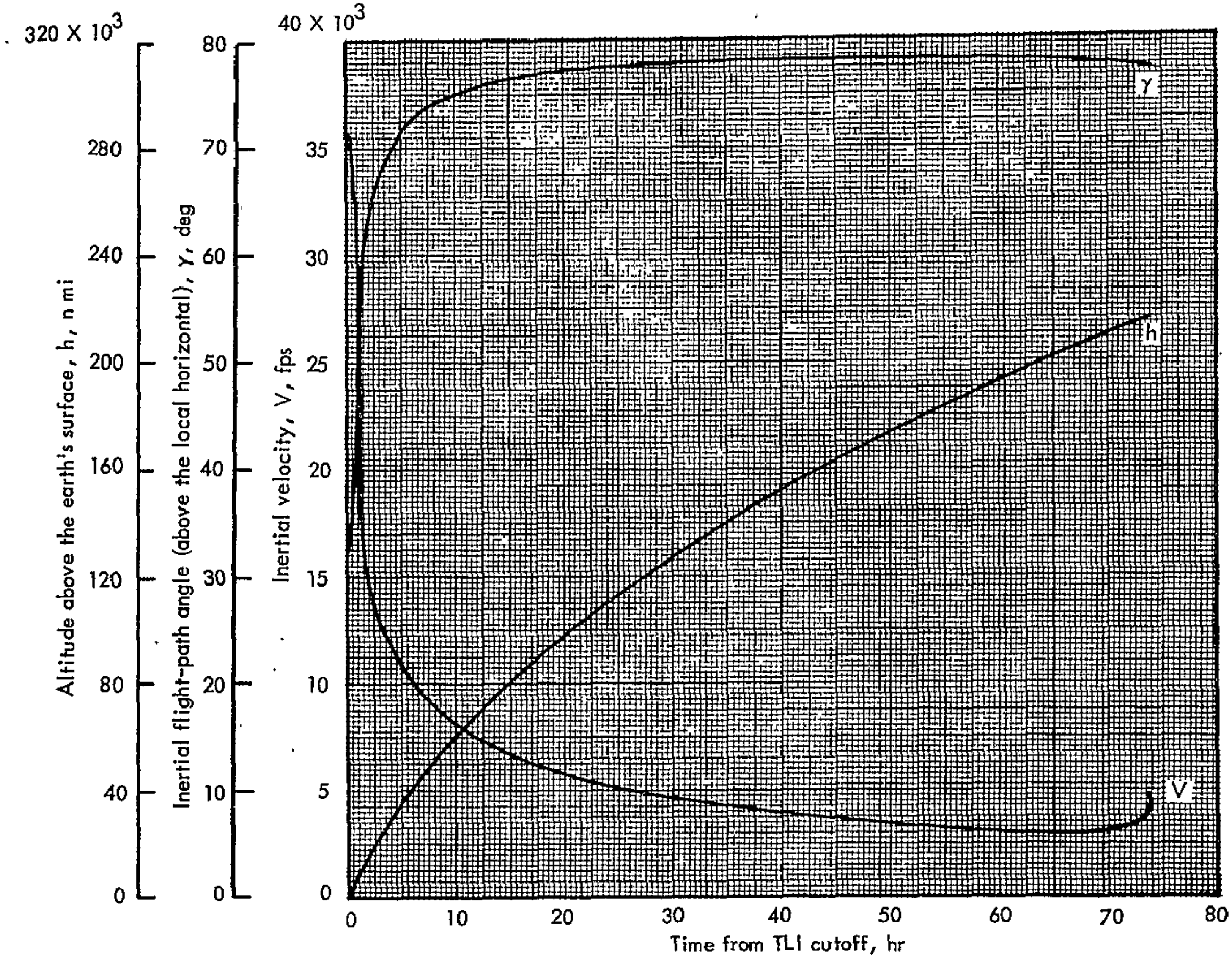


Figure 3.1-1.- Variation of trajectory parameters during translunar coast.

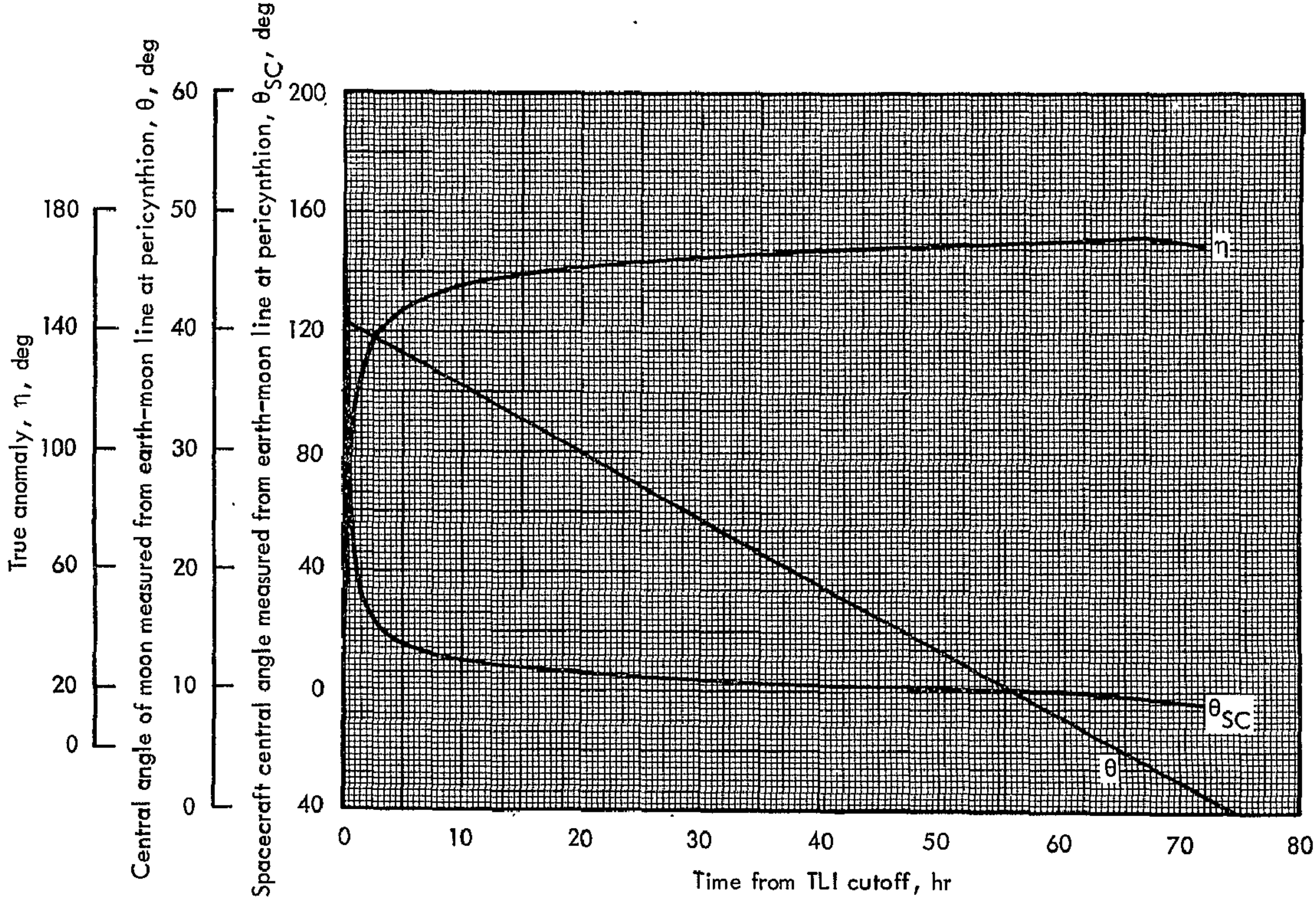


Figure 3.1-2.- Relative positions of earth, moon, and spacecraft during translunar coast

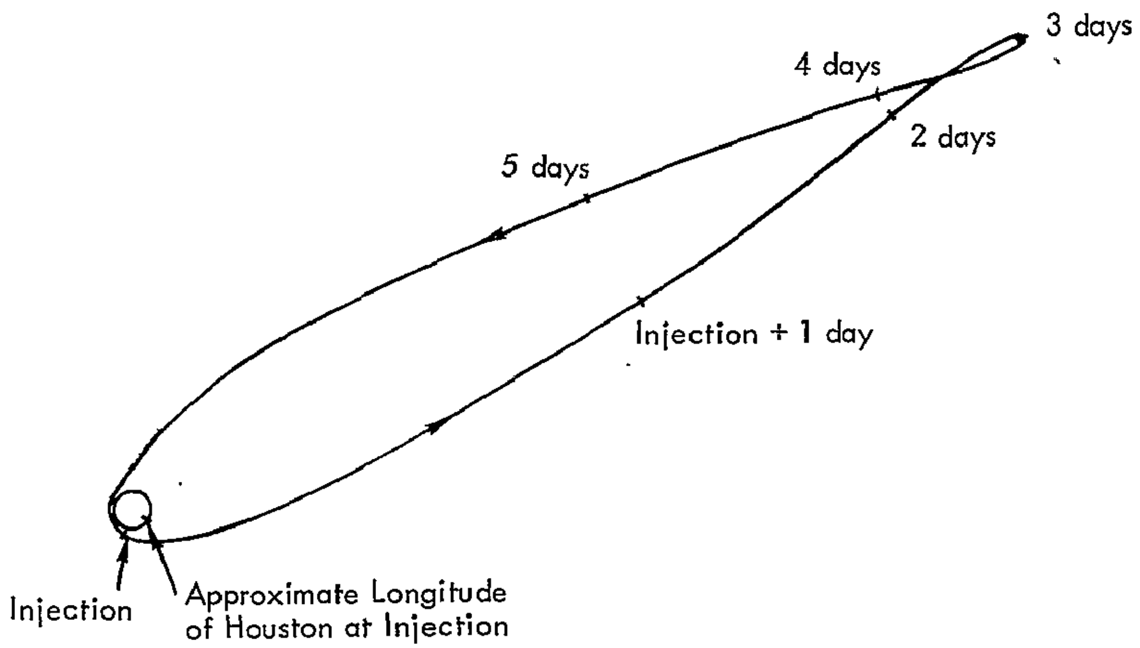


Figure 3.1-3.- Schematic representation of free-return circumlunar trajectory showing approximate time scale.

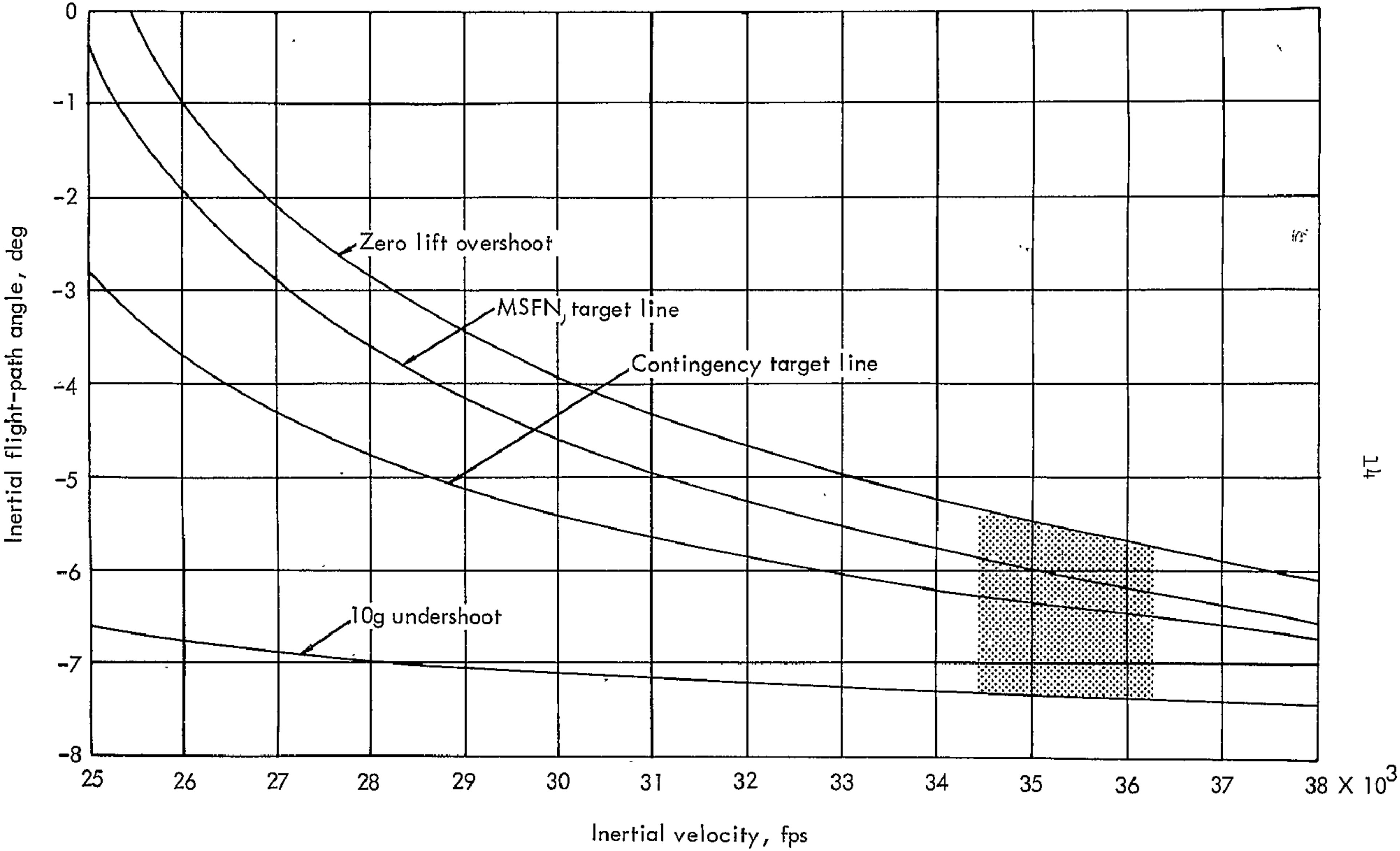
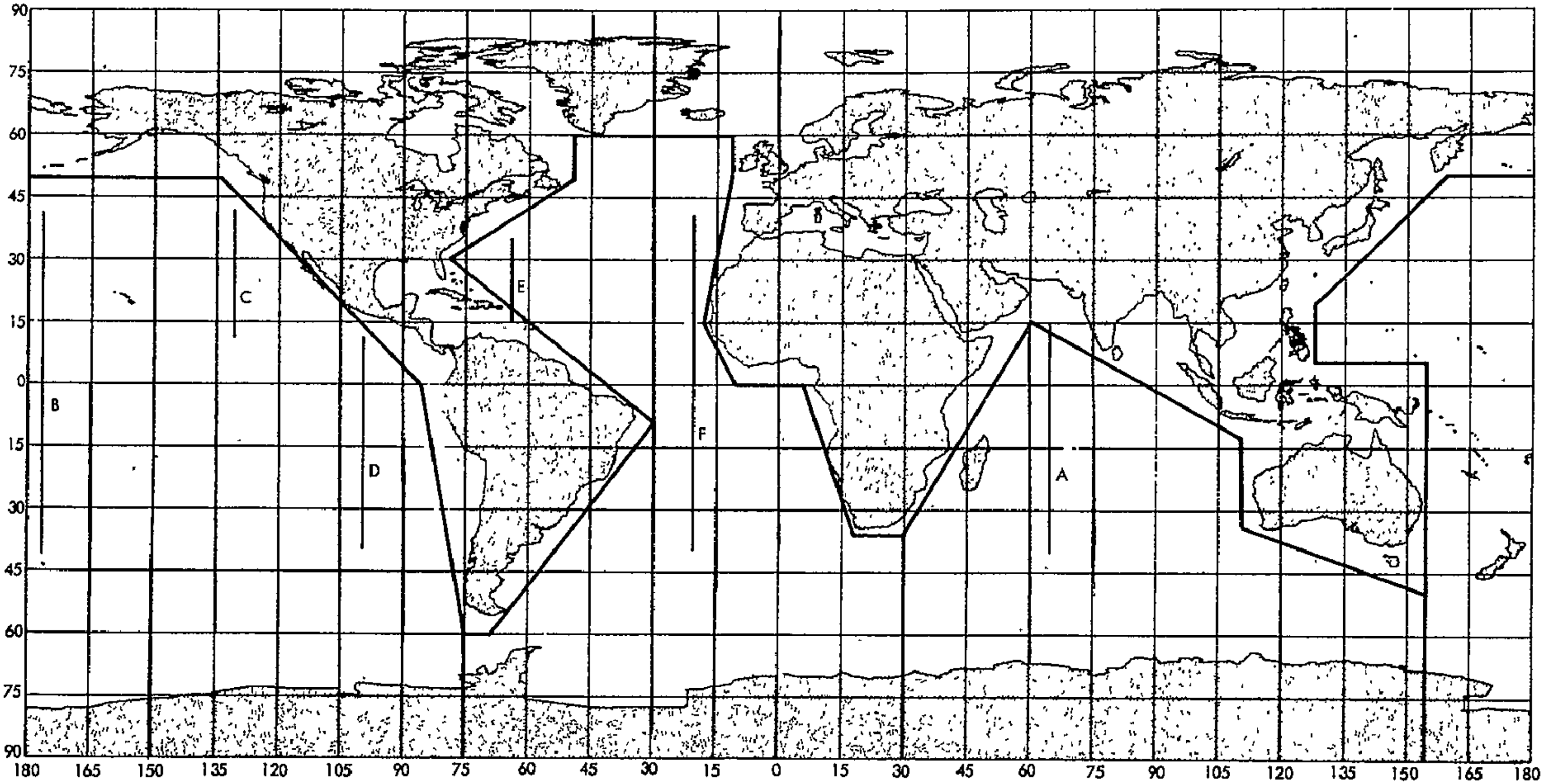


Figure 3.2-1.- Entry corridor and target line definition.



(Reference 14)

Figure 3.3-1.- Land-Water Boundaries and Contingency Lines

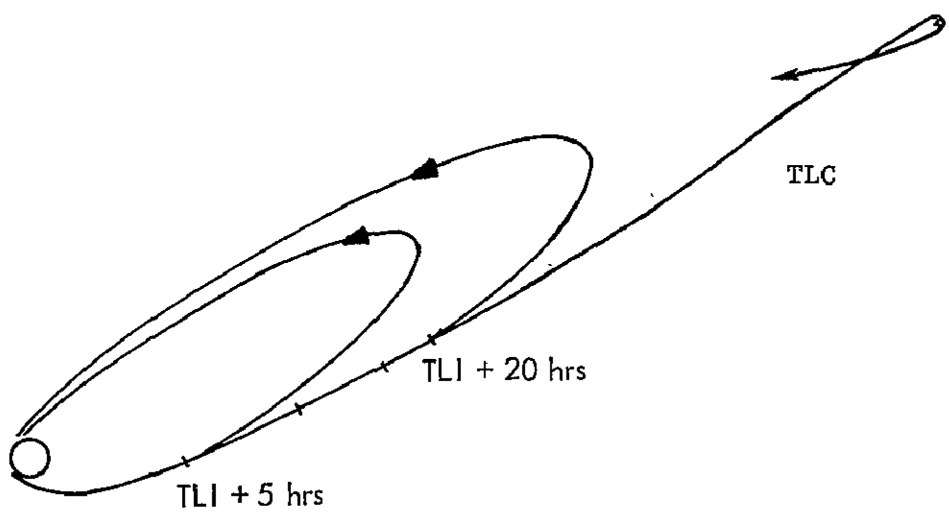
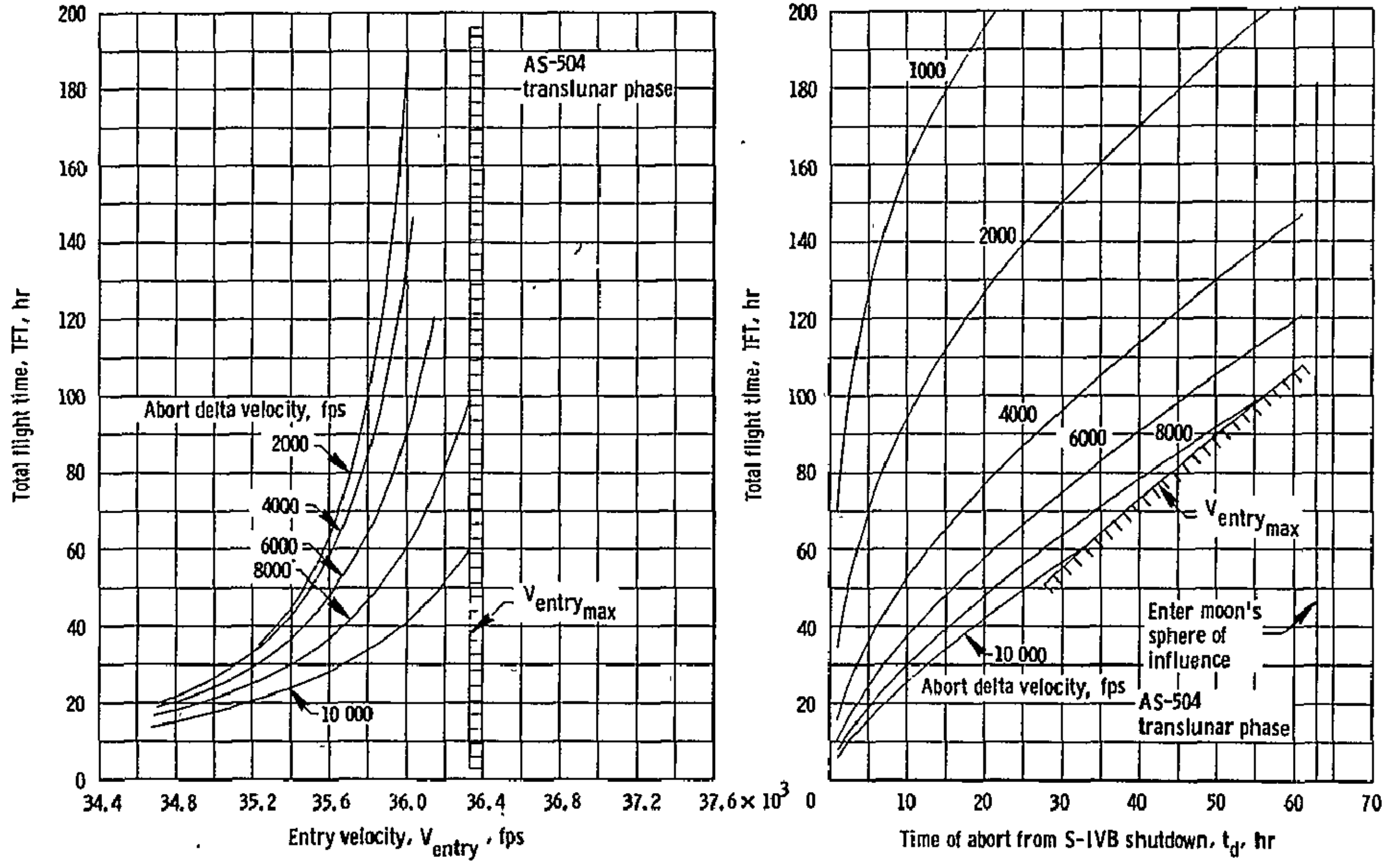


Figure 3.3-2.- Abort trajectories from early translunar coast - with a ΔV of 1900 feet per second.



(Reference 7)

Figure 3.3-3.- Unspecified-area abort analysis during nominal translunar coast.

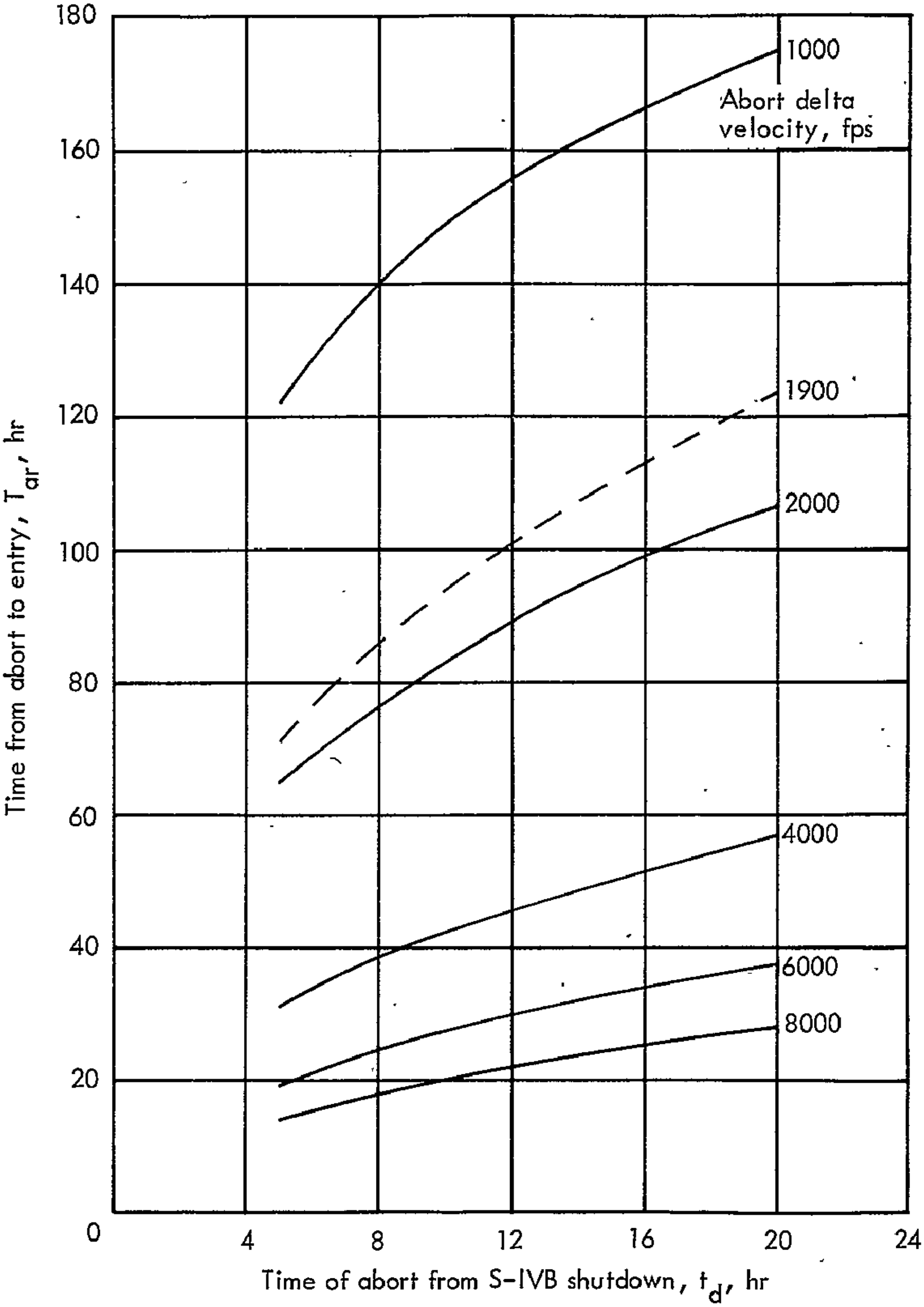
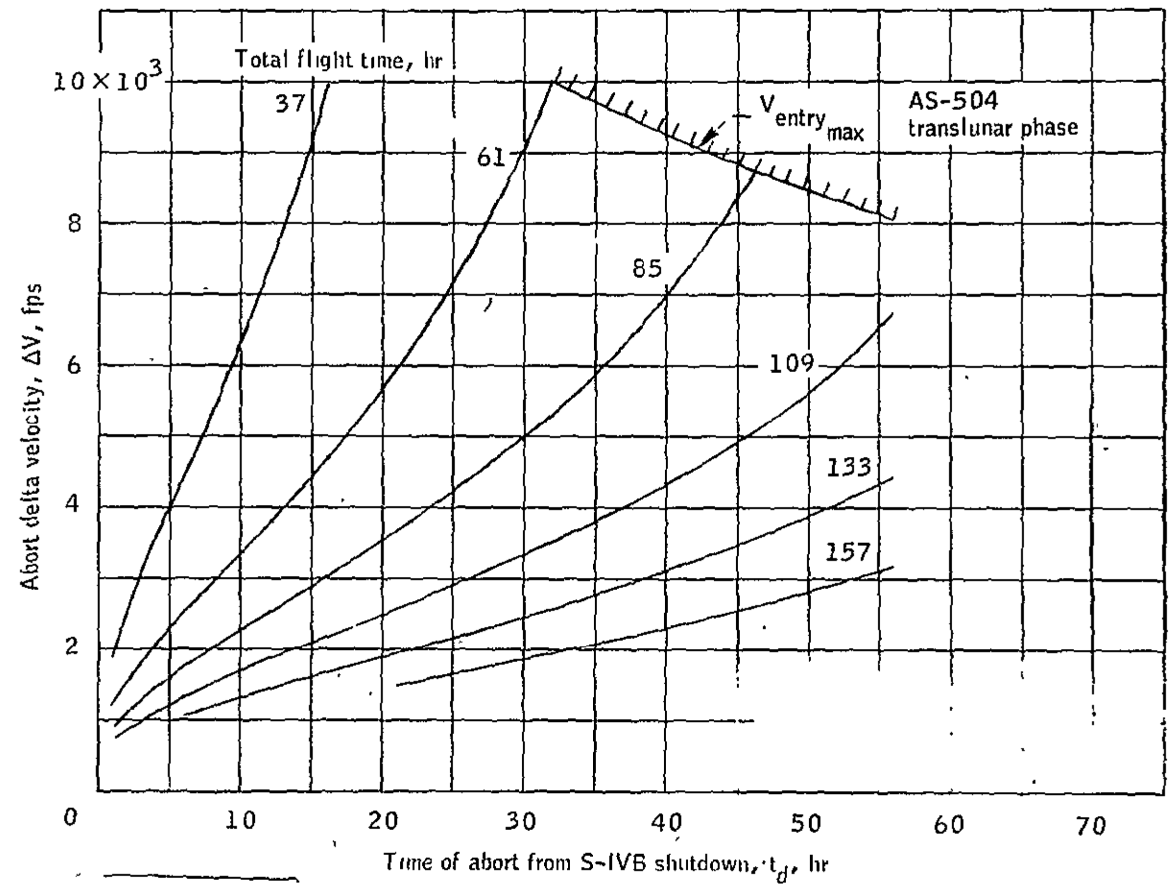
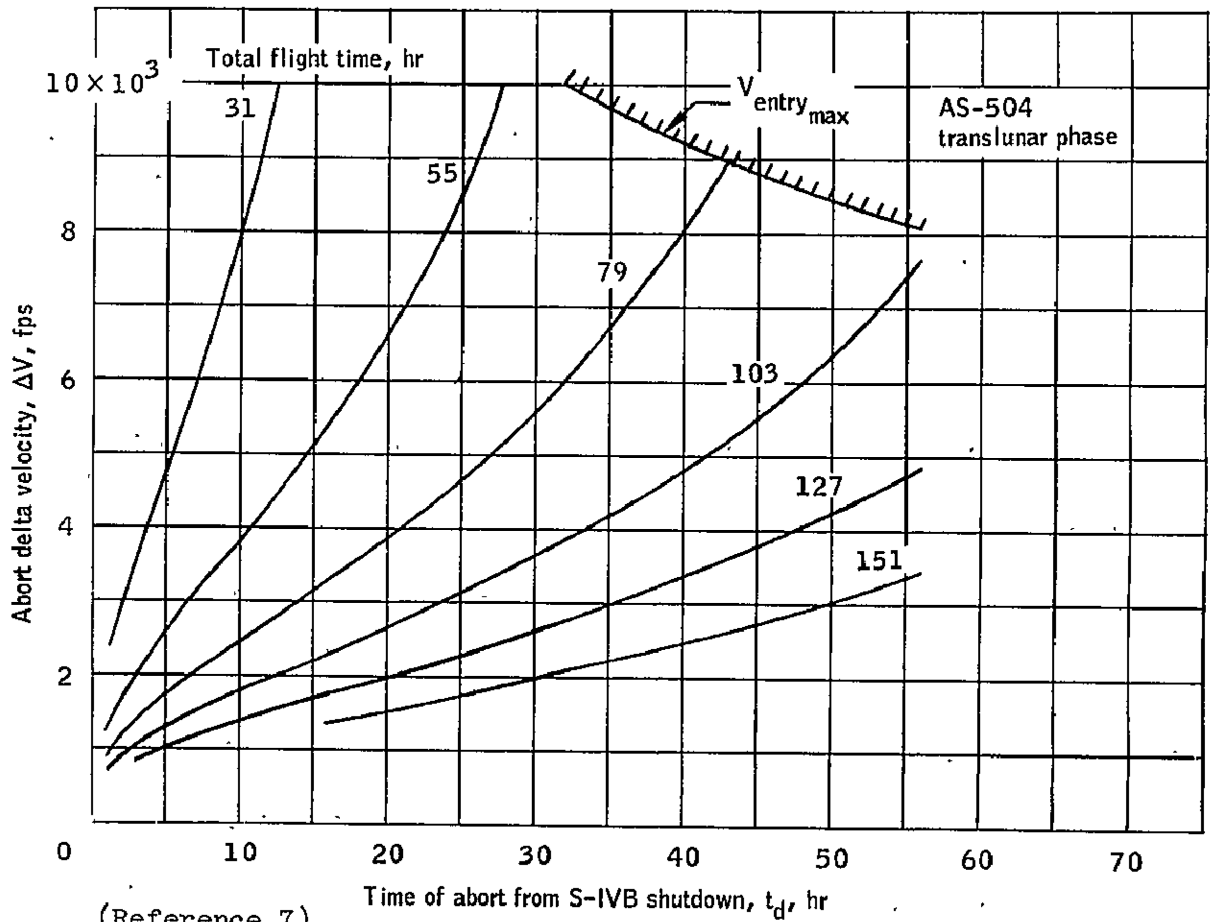


Figure 3.3-4. - Postabort return time versus preabort delay time for unspecified-area aborts from translunar coast.



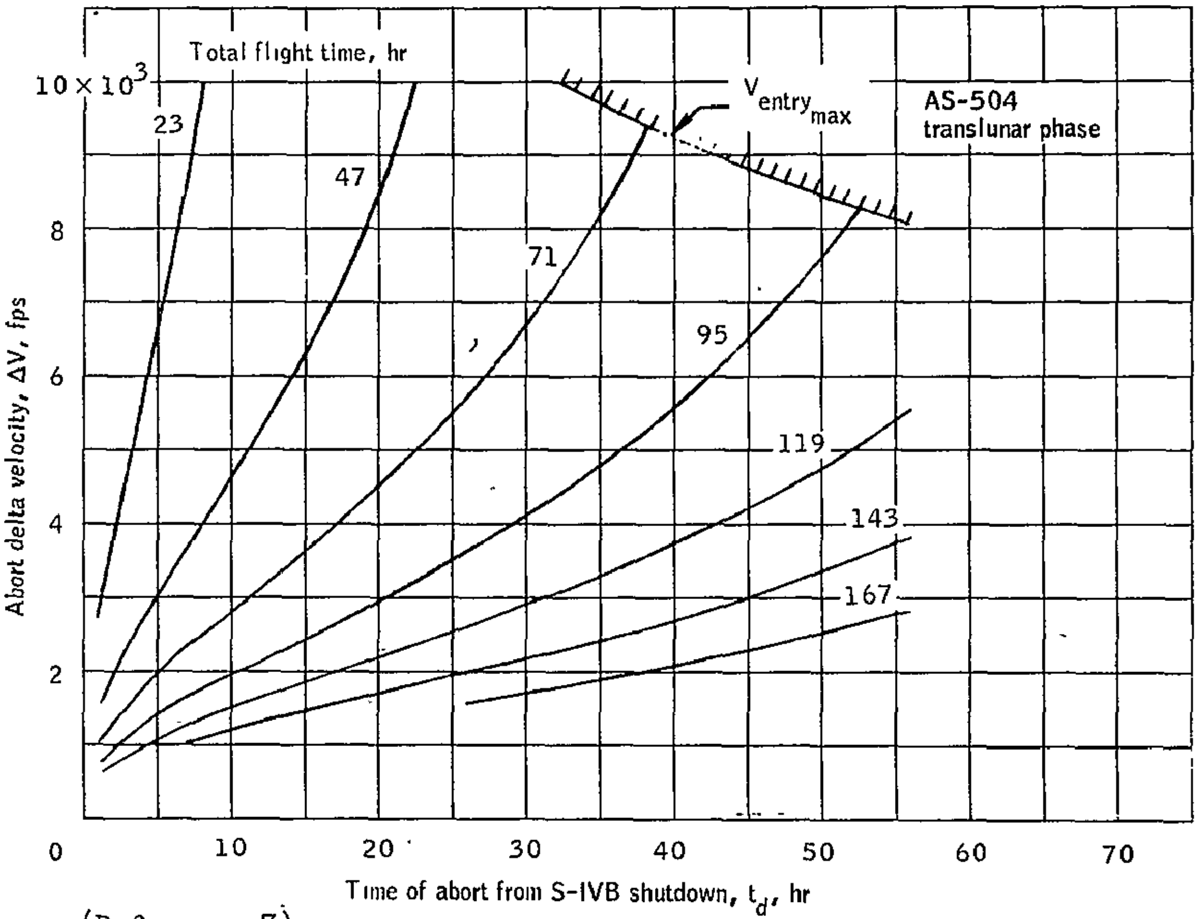
(Reference 7)

Figure 3.3-5. - Abort delta velocity and flight time required for contingency landing area A, CLA A.



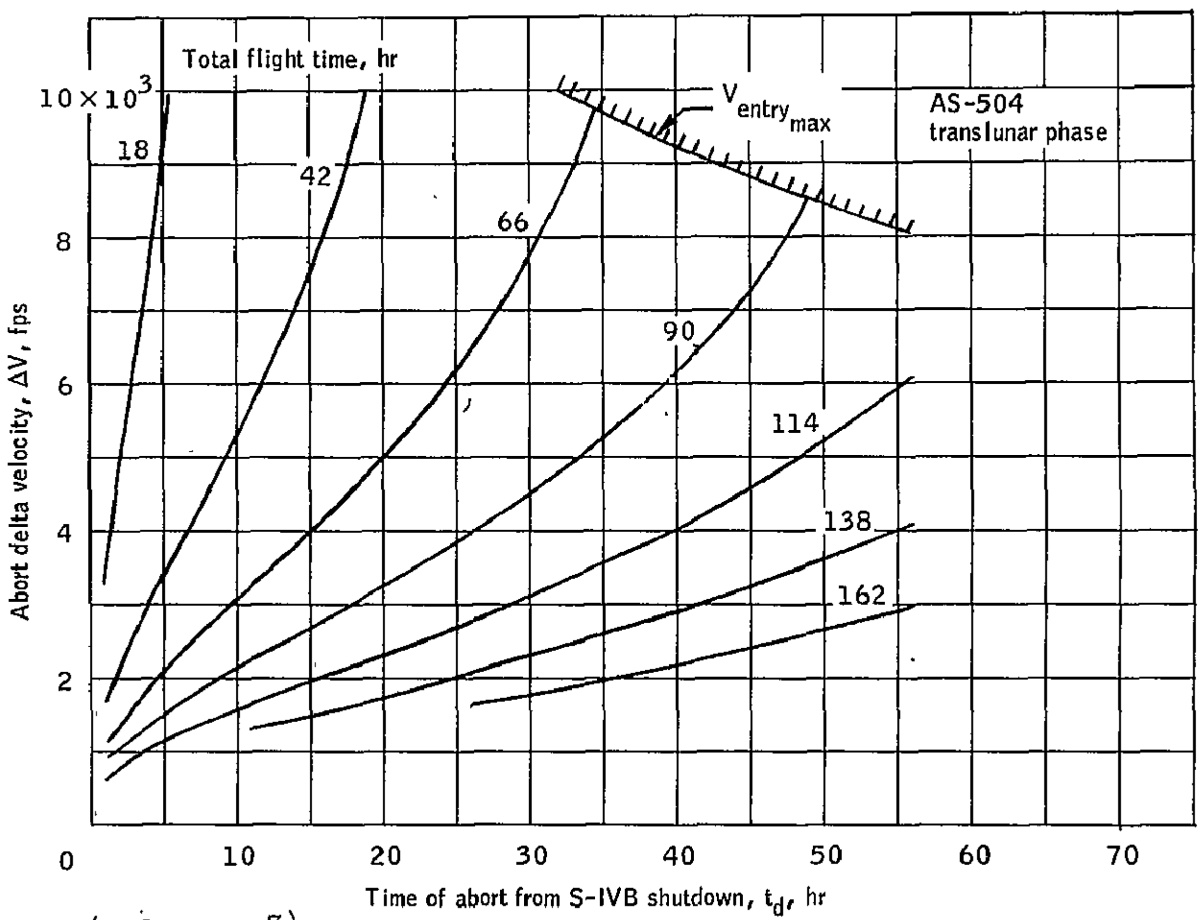
(Reference 7)

Figure 3.3-6. - Abort delta velocity and flight time required for contingency landing area B, CLA B.



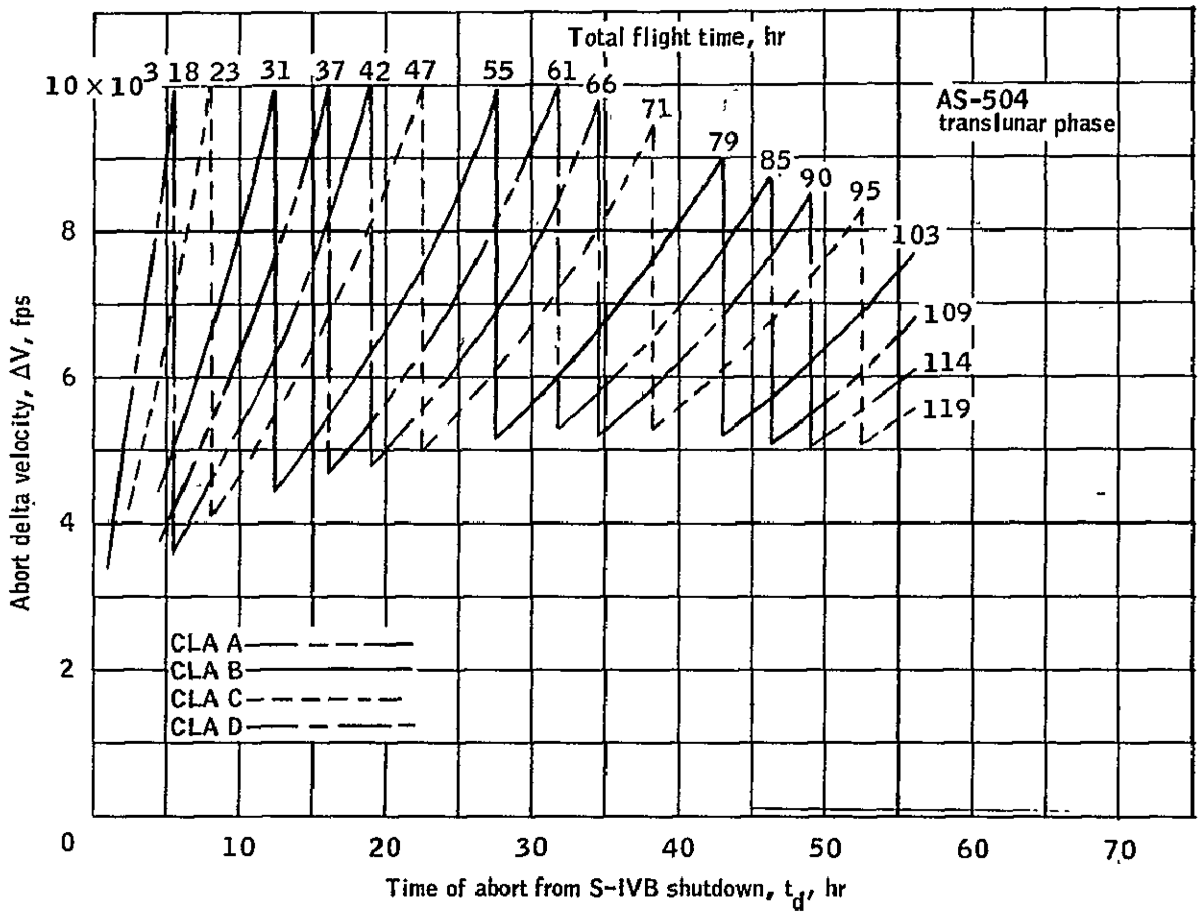
(Reference 7)

Figure 3.3-7. - Abort delta velocity and flight time required for contingency landing area C, CLA C.



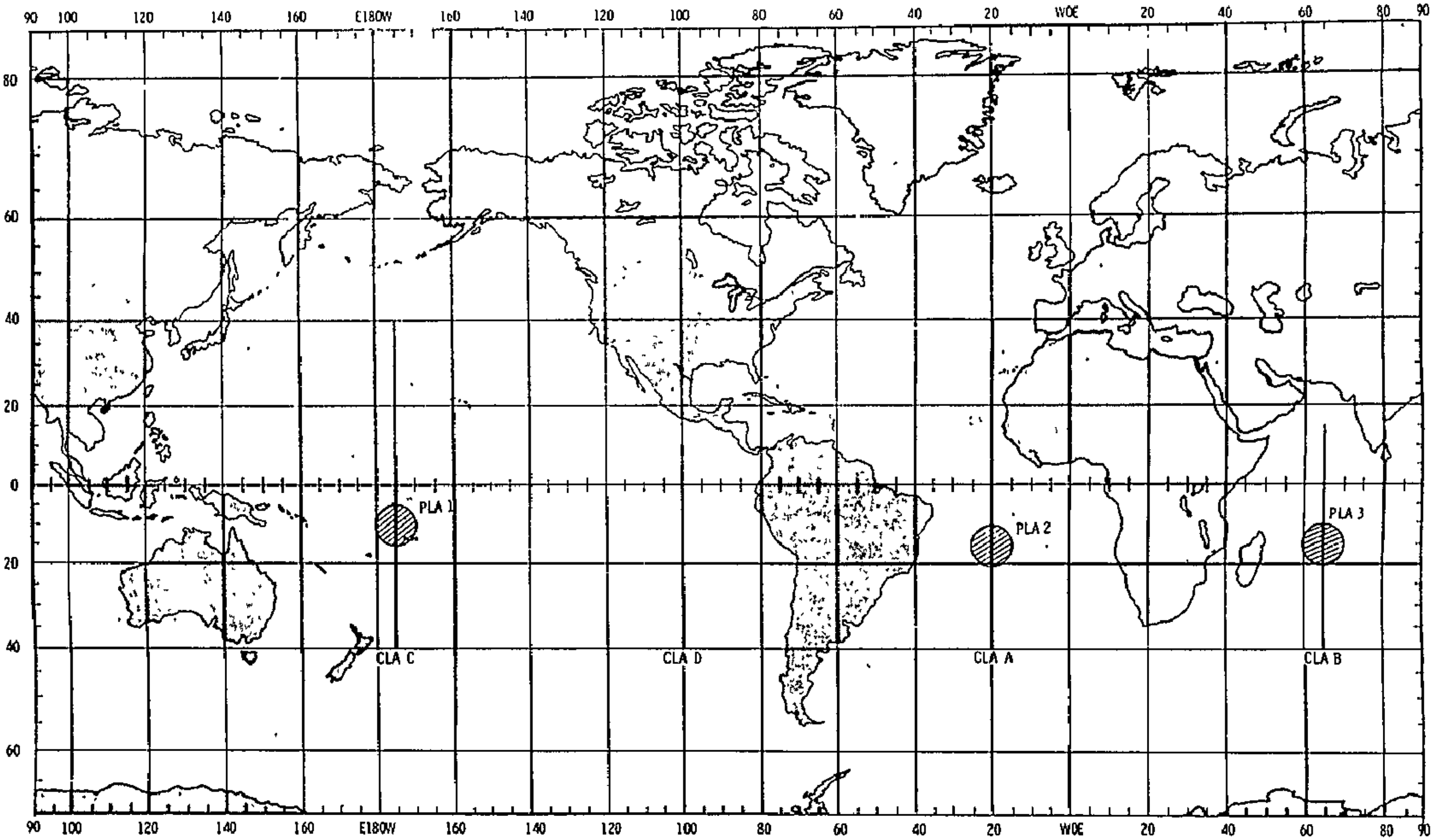
(Reference 7)

Figure 3.3-8. - Abort delta velocity and flight time required for contingency landing area D, CLA D.



(Reference 7)

Figure 3.3-9. - Summary of abort delta velocity and flight time required for minimum time returns to contingency landing area during nominal translunar coast.



(Reference 7)

Figure 3.3-10. - Primary landing areas and contingency landing areas for AS-504.

4. RETURN-TO-EARTH ABORT PROCESSORS

4.1 Real-Time Computer Complex Program

The return-to-earth abort processor (RTEAP) is the real-time computer complex (RTCC) logic that provides return-to-earth trajectories for high ellipse earth orbital and lunar missions. The RTEAP has complete return-to-earth abort capabilities, with the following exceptions. It cannot determine:

1. a TLC midcourse maneuver performed in earth reference

2. a circumlunar abort maneuver performed in earth reference

Analytic trajectories are computed for the abort trade-off display, which presents graphical data used by the flight controller to select an optimum abort solution. This analytic data also provides initial estimates for the impulsive precision solutions generated for the abort scan table (AST) display. The trade-off display is basically a plot of abort ΔV versus time of abort, with multiple curves to indicate different times of landing. For PLA aborts, landing miss distance also can be shown as a parameter. The curves are generated by scanning the desired region of abort time, using a time increment (between solutions) which is determined by the program logic. Furthermore, scale values for the graphs are generated internally by the program, and are functions of the solutions. In the event that a flight controller desires to see a specific region of the graph expanded, this is accomplished by regenerating data for the desired region with a finer scan interval. The necessary changes in scaling are made automatically since, as stated above, the scaling is driven by the abort solutions.

Two options are available in the RTEAP trade-off display, the PLA/CLA near-earth option and the PLA/CLA remote-earth option. The latter option is designed for flight regions where the inertial entry point is approximately fixed, i.e., for earth-focused eccentricities greater than about 0.8. Thus, the remote-earth option is applicable to TLC abort situations. Figure 4.1-1 (taken from Reference 9) is an example of a remote-earth trade-off for PLA data. Trade-off displays of CLA data have the same general form.

The abort scan table is a digital summary of several abort trajectories generated in the following three steps. First, an analytic solution is obtained using an impulsive thrust maneuver, a conic coast, and a curve-fit entry. Next, the generalized iterator program produces an impulsive-precision trajectory by integrating from the postabort state to the entry state. Thirdly, either a curve-fit or a precision entry simulation produces the landing state. The following options are available in the AST:

Time-critical unspecified-area (TCUA)

Fuel-critical unspecified-area (FCUA)

PLA/CLA discrete

PLA/CLA fuel-critical search

PLA/CLA time-critical search.

Aborts generated under the first two of these options are constrained to the preabort plane of motion, subject to the return inclination constraint. As a further restriction, the AST is incapable of generating abort solutions having a specified ΔV alignment with respect to visual references. Thus, horizon reference aborts, for example, could be computed only by manual iteration to achieve the desired abort $\vec{\Delta V}$ alignment. Tables IV and V give the input quantities for the TCUA and FCUA options. The information in these tables is taken from References 10 and 11, which constitute complete documentation of the RTEAP logic.

The final abort solution is a fully integrated trajectory which is generated in the return-to-earth digital (RTED) display. The AST solution is nominally used as input to the RTED, but the option exists to manually input an external solution. Information from the RTED display is transmitted to the command module computer (CMC) for subsequent use in execution of the abort maneuver. Reference 11 gives the complete input required for the RTED display.

4.2 Onboard Return-to-Earth Abort Program

The command module computer (CMC) program 37 (COLOSSUS P37) provides onboard capability for computing a safe earth-return trajectory, providing the command service module (CSM) is outside the moon's sphere of influence (MSI) at the time of ignition. COLOSSUS P37 is primarily a backup in the event of communication or ground system failure and is self-contained in that it operates without the use of the CSM inertial subsystem (ISS) and without ground communication. Minimum-time and minimum-fuel modes are available, with all return trajectories constrained to lie within the preabort plane. The computation includes a conic two-body solution followed by an impulsive-precision integrated trajectory. A complete description of the program can be found in Reference 12.

After calling P37, the astronaut is required to manually input the following information:

1. Delta-velocity to be used in the maneuver, ΔV

2. Time of ignition, t_{ig}

3. Inertial flight path angle at entry, γ_{EI} (if desired to target to any γ_{EI} other than built in target line)

TABLE IV. INPUT QUANTITIES FOR THE AST TIME-CRITICAL UNSPECIFIED-AREA OPTION

Required Quantities

ΔV_{max}	The maximum ΔV to be used for the abort maneuver
$T_{o\ min}$	The time at which the maneuver is to be computed
T_{vect}	The state vector time in the mission plan table (MPT) ephemeris
EP	A flag specifying reentry simulation mode

Optional Quantities

$I_{R\ max}$	Maximum entry inclination
$V_{R\ max}$	Maximum entry speed
RRBIAS	An entry down-range bias added to the value obtained from the entry curve fits
TGLN	A flag selecting the entry target line (if other than the nominal)

TABLE V. INPUT QUANTITIES FOR THE AST FUEL-CRITICAL UNSPECIFIED-AREA OPTION

Required Quantities

$T_{o\ min}$	The minimum time at which maneuvers will be computed
T_{vect}	The state vector time in the mission plan table (MPT) ephemeris
EP	A flag selecting the entry mode

Optional Quantities

$T_{o\ max}$	The maximum abort time
$T_{z\ max}$	The maximum landing time
CIRI	The direction of motion flag for lunar phase aborts (see reference 10)
$h_{pc\ min}$	Minimum pericynthion altitude
I_{Rmax}	Maximum entry inclination
V_{Rmax}	Maximum entry speed
TGLN	A flag selecting the entry target line (if other than the nominal)
RRBIAS	Entry downrange bias added to the entry curve-fit value.

The minimum-fuel option generates a return trajectory satisfying the contingency target line at entry (built into the program), and minimizes the required ΔV. To insure a minimum-fuel solution, the input ΔV should be zero. If the ΔV input is not zero, the program will, if possible, use the indicated value of ΔV to compute a minimum-time solution; i.e., a trajectory satisfying the entry target line and returning as soon as possible. If the input ΔV is insufficient to ensure a safe return, a minimum-fuel solution is obtained. In any event the generation of a minimum-fuel solution is accompanied by an output giving the magnitude of the required ΔV. The value of γ_{EI} is normally input as zero requiring P37 to calculate γ_{EI} internally using the built in target line.

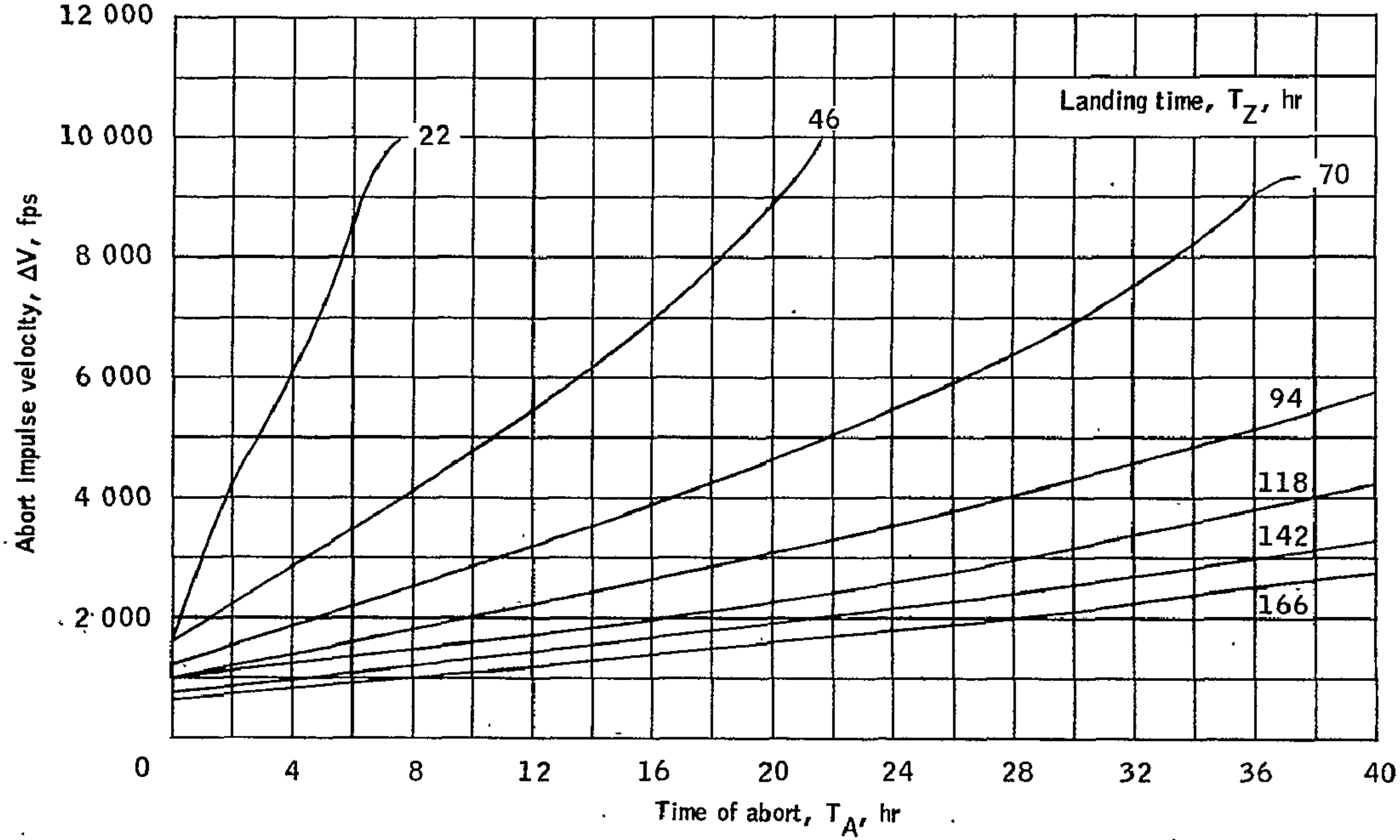
At the completion of the conic trajectory calculation, and again following the precision integrated solution, the program displays values of the following parameters:

1. Landing point latitude
2. Landing point longitude
3. Required $\Delta\vec{V}$ vector
4. Velocity magnitude at 400,000 feet
5. Flight-path angle at 400,000 feet
6. Time from abort ignition to 400,000 feet

When a propulsion system has been chosen for the maneuver, three additional parameters are displayed: middle gimbal angle at ignition, ground elapsed time of ignition (including computed ignition backup time), and time from ignition. The program has a recycle capability by which the astronaut can adjust input quantities and thus exercise control over the longitude of the abort landing site. It should be noted that P37 does not contain built-in constraints on return inclination, entry speed, or minimum or maximum return time. Since all solutions are inplane, no inclination control is available. The program outputs, however, do enable the astronaut to adjust the remaining parameters by manual iteration. The only P37 constraint on the abort point is that although the program will integrate the CSM state vector through the moon's sphere of influence (MSI), it will not reliably compute a return-to-earth trajectory for an ignition time within the MSI. Thus it will generate a postpericynthion abort solution with a t_{ig} following MSI exit.

Execution of P37 causes the storing of return-to-earth target parameters for later use by COLOSSUS P40 or P41, which are service propulsion subsystem (SPS) and reaction control subsystem (RCS) thrusting programs, respectively. The cross product steering constant is set equal to unity for use by the Lambert aim point maneuver guidance.

The appendix is extracted from References 12 and 13 and contains the simplified logic diagram of the return-to-earth targeting, as well as the CMC logic/astronaut checklist interface for COLOSSUS P37.



(Reference 9)

Figure 4.1-1. - Trade-off display PLA/CLA remote-earth option (PLA data, earth centered)

5. SPACECRAFT SYSTEMS AND EQUIPMENT CAPABILITIES

CSM Subsystems

5.1.1 Propulsion.- The service propulsion subsystem (SPS) is the primary thrust system for abort maneuvers executed during the LLM. It is contained in the service module and is nominally used during a lunar mission for lunar, transearth, and orbital maneuvers. The major components of the SPS are a helium pressurization system, a propellant feed system, a propellant gauging and utilization system, and a rocket engine. Service life for the SPS engine is about 750 seconds, and the engine will support 50 starts and shutdowns. The SPS thrust chamber is gimbaled to enable trim alignment of the thrust vector, which passes through the intersection of Y_A and Z_A axes (of the CSM coordinate system) at $X_A = 833.2$ inches (reference 17). At the null offset position, the thrust vector is offset from the X_A axis +0.95 degrees in yaw and -2.15 degrees in pitch. These null offsets compensate for differences between the CSM center of mass and its geometrical center. The gimbaling mechanism can provide ± 4.5 degree inclinations from the null offset position, measured about the Y_A (pitch motion) and Z_A (yaw motion) axes.

Propellant and burn time requirements for ΔV maneuvers are functions of spacecraft weight, engine performance parameters, and the desired ΔV. Nominal block II SPS thrust is 20,000 pounds, resulting from a specific impulse of 315.2 seconds and a mass flow rate of 63.5 pounds mass per second. Figures 5.1-1 and 5.1-2, taken from reference 18, are nomographs of block II SPS performance for the CSM and the CSM/LM configurations, respectively. Two examples of the use of the nomographs are shown in figure 5.1-2. A line joining $W_{SC} = 96{,}000$ pounds (spacecraft weight) with $\Delta V = 1900$ fps intersects the propellant weight/burn time axis at 16,400 pounds and about 258 seconds, thus showing the maneuver expendables for this specific ΔV. For the same spacecraft configurations (same W_{SC}), the maximum available ΔV (5450 fps) is found by drawing a line through an assumed maximum propellant weight of 40,000 pounds.

The service module reaction control subsystem (SM RCS) consists of four clusters (quads) of four thrust chambers each, located 90 degrees apart around the forward portion of the SM periphery. Each RCS quad incorporates a pulse-modulated, bipropellant system to provide thrust for separation, docking, propellant-settling (for the SPS), and attitude maneuvers. Although nominal operations of the SM RCS are in a pulsed mode, the performance parameters for a single RCS jet are given in terms of steady-state operation. For a single jet the specific impulse is 276.0 seconds and the propellant flow rate is 0.361 pounds per second, yielding a thrust of 99.8 pounds.

As a backup system the SM RCS is extremely ΔV limited. At a spacecraft weight of about 96,000 pounds (loaded CSM/LM), four RCS jets firing (in the same direction) for an assumed maximum operating time of 500 seconds provide an impulse velocity of about 67 fps. An equivalent burn results in a ΔV of approximately 300 fps when applied to an empty CSM without the LM attached. Nominal SM RCS translational maneuvers; i.e., SPS propellant settling and small trajectory adjustments, involve ΔV values less than 10 fps. Performance characteristics in this ΔV range are shown in figure 5.1-3, taken from reference 18.

5.1.2 Thrust vector alignment.- Alignment of the SPS thrust vector for a ΔV maneuver, e.g. an abort maneuver, consists essentially of two steps. First, the spacecraft is maneuvered to the desired thrusting attitude supplied from the ground or by the CMC pre-thrusting program. Once this attitude has been established, the SPS engine bell is trimmed in the pitch and yaw planes such that the thrust vector passes through the current spacecraft center of mass. Nominally both of these steps are performed automatically, but manual capability is provided for those cases where inoperable subsystems require it.

In the primary guidance, navigation and control subsystem (PGNCS) automatic mode, thrust vector alignment may be accomplished as follows. Each of the pre-thrusting programs of the CMC (COLOSSUS P30, P31, and P37) computes and stores the components of the ΔV vector for the desired thrusting maneuver, as well as the initial thrust direction. These programs also display the middle gimbal angle (MGA) that would result if the maneuver were performed with the existing inertial measurement unit (IMU) orientation. Should the resulting MGA be larger than 45 degrees, gimbal lock and loss of inertial reference is likely. In this case it is desirable to realign the IMU stable member to a preferred alignment for the impending maneuver; i.e., arbitrarily force the stable member (SM) axes to the following directions:

$$\vec{X}_{SM} = \text{UNIT } (\vec{X}_B)$$

$$\vec{Y}_{SM} = \text{UNIT } (\vec{X}_{SM} \times \vec{r})$$

$$\vec{Z}_{SM} = \vec{X}_{SM} \times \vec{Y}_{SM}$$

where $\vec{X}_B$ is the vehicle X-axis at the preferred vehicle attitude for ignition and $\vec{r}$ is the spacecraft position vector at ignition. This realignment is accomplished by calling P52, a program which provides the option of realigning the IMU to any one of four orientations, including the preferred orientation. (The astronaut may at this point choose to manually maneuver the spacecraft to the desired burn attitude. Following execution of P52 with the preferred alignment option, this is simply a matter of maneuvering so as to zero the angles on the flight director's attitude indicator (FDAI) ball). Prior to use of the SPS thrusting program (P40), the CMC digital autopilot (DAP) loading routine, COLOSSUS R03, must be executed. In routine 03 the DAP spacecraft configuration and weights are verified (updated by astronaut if incorrect), and the SPS engine trim angles are calculated and displayed for verification. Upon completing the DAP routine, the astronaut keys in verb 46 ENTER to establish automatic guidance and navigation control, which includes activation of the appropriate section of the DAP. The next step is selection of P40, the SPS thrusting program, for the monitoring of the thrusting maneuver. One of the early steps of this program is the calling of CMC routine 60, which automatically maneuvers the spacecraft to the initial thrust attitude, by use of the DAP. Finally the pitch and yaw trim maneuvers are executed -- also under automatic control of the DAP -- and the ΔV thrust alignment is complete.

If the PGNCS is inoperative, some of the above operations must be performed manually by the crew with ground assistance. To establish reference the pilot will sight on a star (or other visual object) designated by ground control. Visual acquisition can be made with the sextant (SXT), the scanning telescope (SCT), or the crew optical alignment sight (COAS). The SXT is a 28-power, dual line-of-sight instrument, with each line of sight having a 1.8-degree field of view. The SCT is of unit optical power, but has a 60-degree field of view. Instrument limitations restrict the effective optical coverage of the SXT and SCT to a 45-degree half-cone centered about the optical shaft drive axis. The COAS is a lighted collimator device with a reticle consisting of a 10-degree circle, vertical and horizontal crosshairs with 1-degree marks, and an elevation scale. Upon acquiring the star, the pilot will uncage one set of the body-mounted attitude gyros (BMAGS), thus changing its function from that of an attitude rate indicator to an inertial reference indicator. The BMAGS are part of the stablization and control subsystem (SCS), which is a backup to the PGNCS. Inertial drift rate for the BMAG system is several orders of magnitude greater than that of the IMU. After establishing an inertial reference with the BMAGS, the pilot can reorient the vehicle to the desired thrust attitude using pitch and yaw angles supplied by ground control. An alternative method for establishing attitude reference involves observation of the earth's terminator. By obtaining a pre-determined attitude with respect to the earth's terminator at a specific time, the crew enables ground control to compute their present inertial attitude, and thus the required pitch and yaw maneuvers to reach the proper ΔV attitude. Earth terminator alignment is discussed further in reference 18. With flight charts of SPS trim angles versus weight, or information supplied from the ground, the SPS engine can be trimmed through the vehicle center of mass. These small maneuvers are performed with the SM RCS under astronaut control, by manual input to the trim wheels.

5.1.3 Guidance.- After a ΔV maneuver has been computed and the spacecraft has been aligned to the initial thrust attitude, it is the function of a guidance system to properly direct the thrust vector for the duration of the burn, insuring the targeted objectives of the maneuver. Powered-flight guidance may be accomplished in one of several ways, as discussed in the following paragraphs.

With all spacecraft subsystems operable, guidance is under PGNCS control and is automatic. Pre-thrusting maneuvers, active engine on/off commands, and continuous thrust-vector control are provided by the CMC DAP for SPS maneuvers (using COLOSSUS P40). There are two available submodes, to accomodate Lambert aim point maneuvers and external ΔV (fixed inertial attitude) maneuvers. In either case, cross-product steering (reference 14) is employed with a basic guidance cycle time of 2 seconds. Each computation cycle is triggered by reading of the pulsed integrating pendulous accelerometer (PIPA) ΔV counters, and contains the following steps.

1. Updating of the spacecraft state vector in the average-g subroutine, including effects of thrust acceleration and gravity
2. Computation of the time to go (t_{go}) before engine shut-off
3. Generation of steering commands for the DAP; i.e., an attitude rate (ω_{CNB}) tending to satisfy the cross-product steering law
4. Updating of the velocity to be gained vector ($\vec{V}_g$) by solution of the Lambert intercept problem (This step is omitted in the external ΔV submode.)

In step 4 above, the magnitude and direction of $\vec{V}_g$ are changed, resulting in rotation of the thrust vector during the burn. When external ΔV guidance is being used, $\vec{V}_g$ is calculated only one time and subsequent steering commands to the DAP serve only to maintain the thrust at a constant inertial attitude. In both guidance submodes the cyclic process of PIPA ΔV measurement, steering computations, and DAP rate commands continues until the value of t_{go} reaches zero, when the engine is automatically commanded off. The engine-off command actually results when $\vec{V}_g$ reaches zero, since t_{go} is proportional to $|\vec{V}_g|$. A complete discussion of PGNCS guidance can be found in reference 12. Figure 5.1-4, taken from this reference, shows the guidance seqqence for a Lambert aim point maneuver.

If the IMU is not on or is not aligned, or if for any other reason the primary system cannot be used, a ΔV maneuver can be performed with SCS thrust vector control (TVC). Such maneuvers are limited to the fixed inertial attitude mode and the spacecraft must be manually oriented to the desired thrusting attitude. Control of this type of maneuver consists only of holding the thrust vector inertially fixed (SCS auto), which is the function of the SCS trim follower integrator. In addition to controlling the direction of $\vec{\Delta V}$, the SCS also commands the SPS engine off when the entry monitoring system (EMS) ΔV counter has counted down to zero. The EMS ΔV counter must be manually set at the desired value prior to an SCS maneuver.

In severe contingencies the abort maneuver may have to be controlled manually. Without ground communication, a $\vec{\Delta V}$ attitude must be determined by astronaut reference to the earth's horizon or terminator, or to the moon's horizon or terminator. In addition, the astronaut must maintain this attitude during the burn by continuous visual observation of his out-of-the-window reference. For a constant-attitude, inplane maneuver, the procedure involves:

1. Determining the orbital plane and aligning the spacecraft's pitch plane (X-Z plane) with it

2. Lining up an appropriate horizontal line of the COAS reticle with (for example) the far earth horizon

3. With the rotational hand controller (of the SM RCS jets), maintaining the required pitch angle with respect to the horizon

4. Manually commanding thrust on and off, assuming the ΔV (as well as horizon reference angle) has been determined from prepared crew charts

5.2 LM Subsystems

5.2.1 Descent Propulsion System (DPS).- The LM DPS is the prime backup system in case of SPS engine failures for TLC aborts. Accordingly, it is appropriate to describe the DPS in terms of the major operational differences between the two propulsion systems.

The DPS is physically smaller and carries less fuel, about 18,000 pounds compared with 40,000 pounds in the SPS. In order to provide hover and lunar soft-landing capability, the DPS engine is throttleable between 65% and 10% of its full throttle point (FTP) thrust of 10,500 pounds. This FTP thrust is approximately half the constant 20,000 pound thrust level of the SPS. Thrust level uncertainties for the DPS are relatively large due to variable erosion of the engine throat.

The fuel-pressurization system for the DPS contains supercritical helium (SHe), which will be loaded about 18 hours prior to nominal liftoff. Due to transfer of heat from the surroundings, the SHe tank pressure increases by about 10 psia per hour until the DPS engine is fired. Safe operation of the engine at FTP requires a SHe pressure range from 450 to 1400 psia, which corresponds to a time range from 37 to 132 hours following SHe loading. The disadvantages of accommodating DPS maneuvers (especially abort maneuvers) to the helium pressure timeline are offset by the obvious advantage of accomplishing fuel pressurization with a lighter, more compact system.

The start sequence for the DPS engine requires that the throttle be held at 10% of full thrust for at least 15 seconds. Under the control of a LM guidance computer (LGC) thrusting program, the 10% period is 26 seconds. This low-thrust period is required for the engine gimbal, which is a slow-response mechanism, to center the thrust vector through the center of mass.

Instantaneous high-thrust operation of the DPS could (with an initial gimbal offset) produce large torques about the center of mass and require excessive LM RCS fuel to maintain stability.

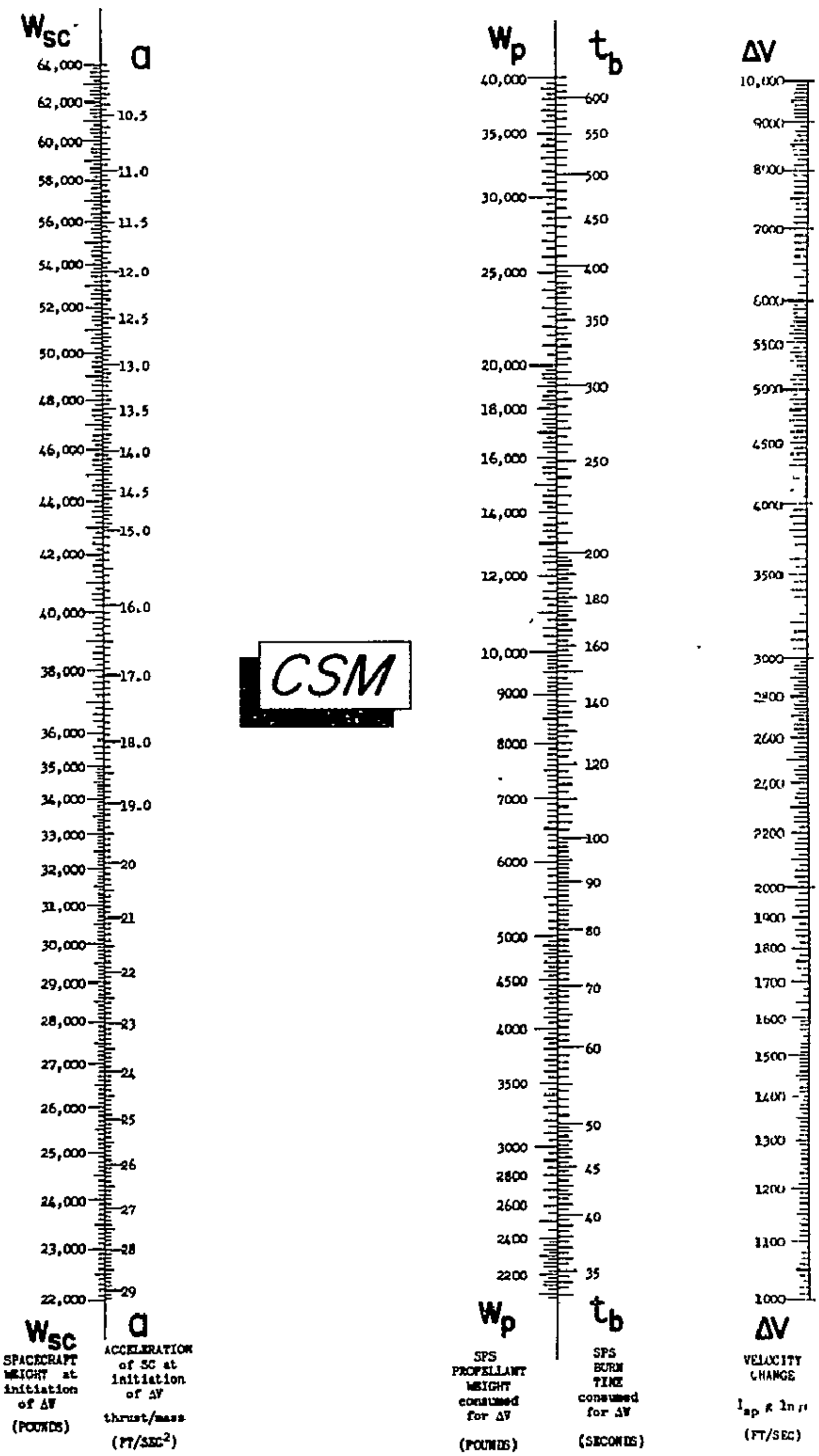
In the event of an SPS failure on an early TLC abort maneuver, the LM DPS could provide a ΔV of 1900 fps to the loaded CSM/LM, by using full-thrust for about 590 seconds.

5.2.2 Thrust Vector Alignment.- The establishment of a $\vec{\Delta V}$ attitude under LM primary guidance and navigation control is similar to the procedure used for SPS maneuvers. LUMINARY program 40 in the LM guidance computer (LGC), contains logic for control of DPS burns (corresponding to COLOSSUS P40). Prior to calling P40, routine 03 is executed in order to verify vehicle weight and inertia data in the DAP. Routine 03 in the LGC also computes and (upon astronaut option) performs the DPS engine trim maneuvers. Following routine 03, verb 46 ENTER is used to activate the DAP. Program 40 includes the execution of routine 60, which maneuvers to the initial thrust attitude specified by the pre-thrusting program (LUMINARY P30 or P31).

Without the primary LM inertial subsystem (IMU), $\vec{\Delta V}$ alignment must be obtained manually using optical measurements to establish inertial reference. This is a difficult procedure because of mechanical limitations on the LM alignment optical telescope (AOT). If the spacecraft can be placed in the plane of two known stars, attitude reference can be established with the aid of ground control, and gimbal settings for the desired maneuver attitude can be supplied.

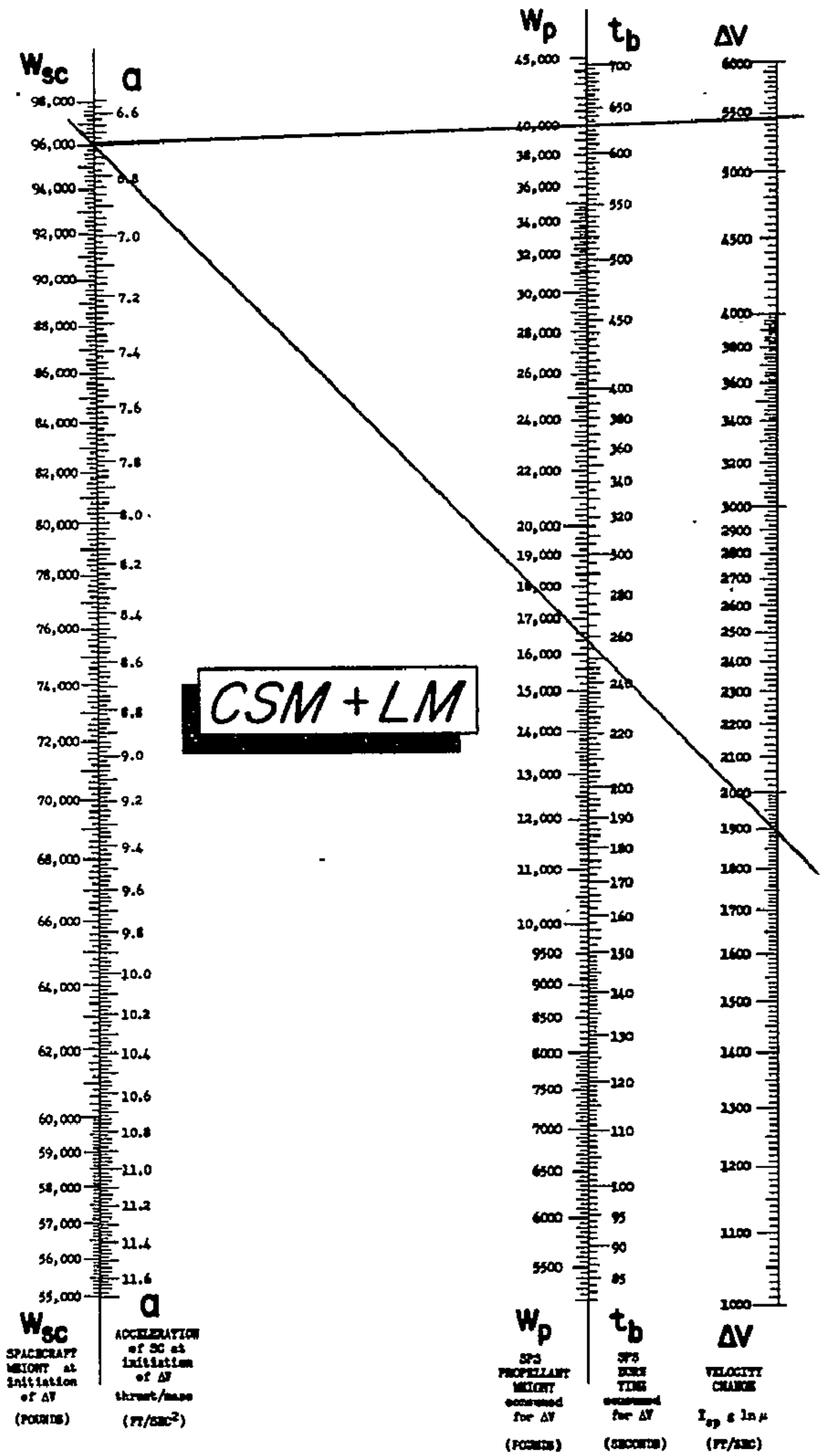
The above considerations refer to the LM only. For abort situations arising after the post-TLI malfunction checks, the LM will be in a docked configuration with the CSM. The use of the LM DPS as a thrust backup to the SPS presents no particular problems of $\vec{\Delta V}$ alignment, except in the extreme case of complete loss of voice communications in both vehicles. In such a situation a block-data abort solution or abort parameters generated by the CMC return-to-earth program (CMC P37) could be manually input to LGC program 30, and the maneuver subsequently performed with LGC program 40.

5.2.3 LGC Guidance.- Powered-flight guidance with LGC program 40 provides the same two modes as CMC program 40: Lambert aim point guidance and external ΔV (constant inertial attitude) guidance. The LGC guidance computation sequence is described in detail in reference 20, and is virtually identical to that in the CMC program 40 (see section 5.1.3). The LM PGNCS also provides an attitude-hold mode which can be used for a ΔV maneuver in a CSM/LM docked configuration.



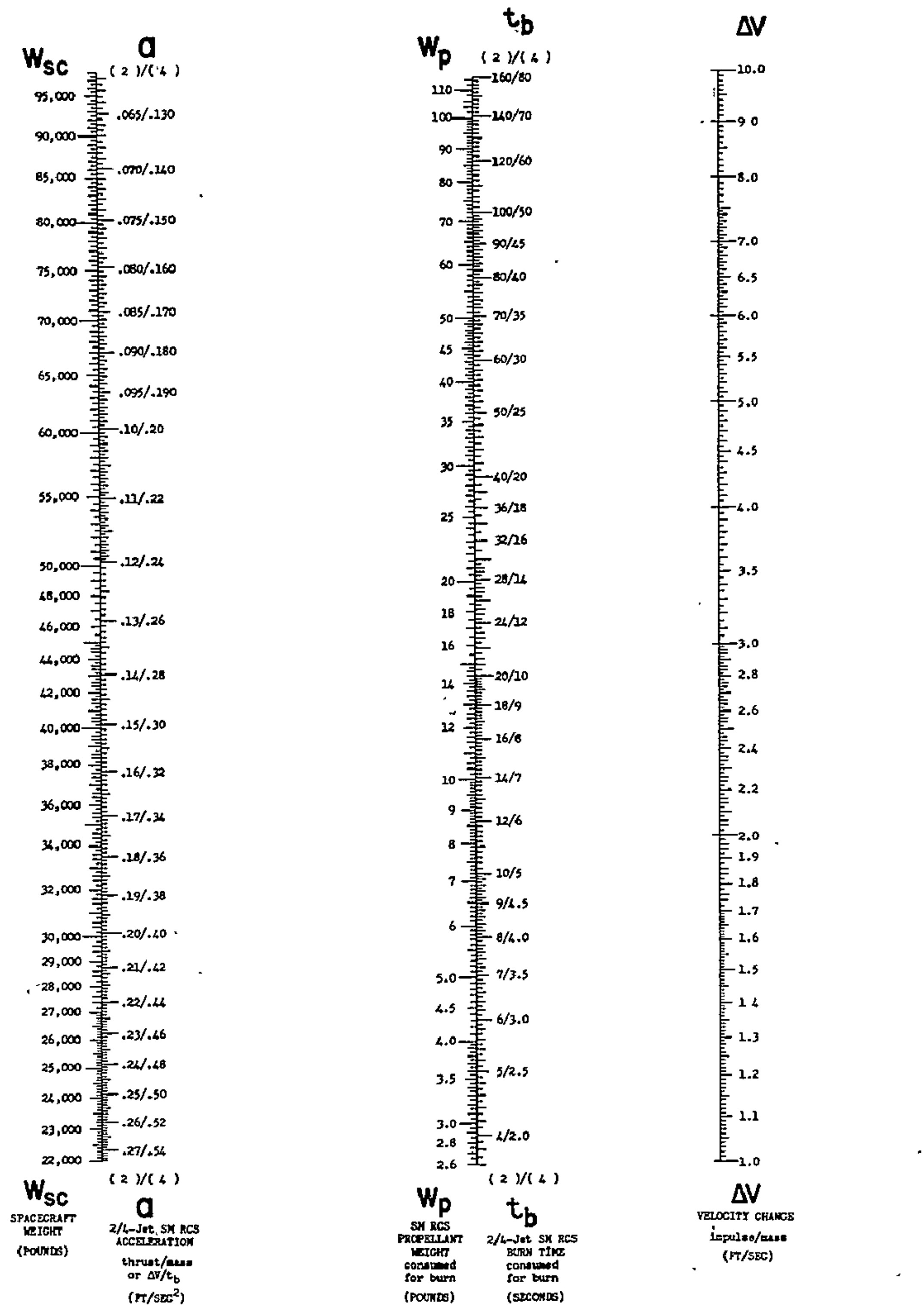
(Reference 18)

Figure 5.1-1. SPS performance nomograph (CSM)



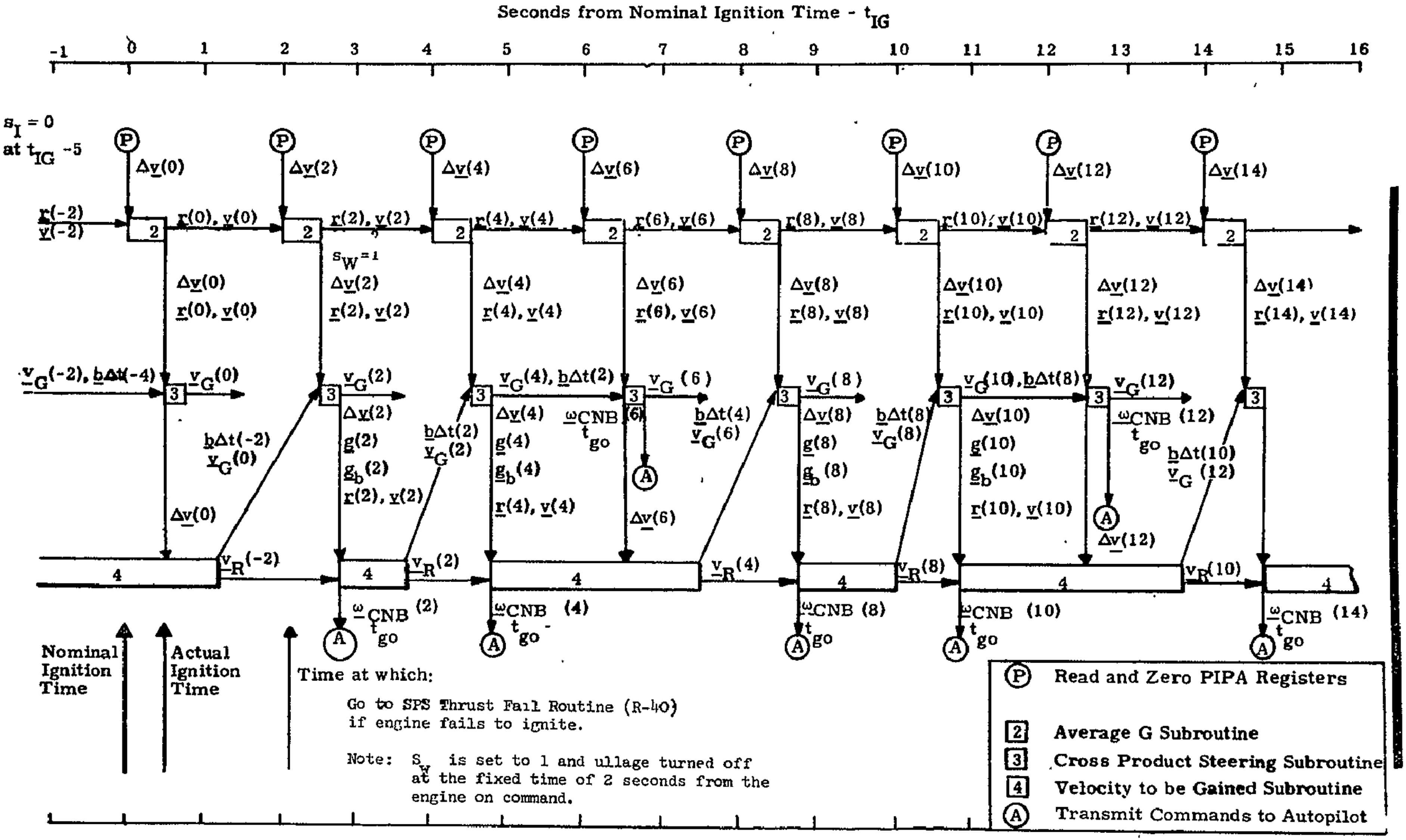
(Reference 18)

Figure 5.1-2. SPS performance nomograph (CSM/LM)



(Reference 18)

Figure 5.1-3. Service module RCS translation performance nomograph



(Reference 12)

Figure 5.1-4. Normal Lambert Aim Point Subroutine Sequencing (Maneuver time greater than 6 sec) for SPS Maneuvers

6. TRANSLUNAR COAST ABORT PROCEDURES

6.1 Ground Rules

Following is a list of ground rules to be used in the establishment of TLC abort procedures. It should be emphasized that these are not mission rules, but do constitute the current rationale for the eventual formation of mission rules concerning abort procedures. These ground rules, as well as the procedural flow charts to follow, will be reviewed initially by the Apollo Abort Working Group.

1. The SPS is the prime propulsion system for all abort situations, and the DPS will be used only to backup an SPS failure.

2. Initial abort attempts will use the SPS in the CSM/LM docked configuration, retaining the DPS for its SPS backup capability.

3. Minimum time for abort maneuver preparation when using the LM DPS is assumed to be 2 hours.

4. If possible the LM should be kept as a communications backup, to provide entry data as close as possible to entry interface.

5. Upon total loss of ground/spacecraft communications, an abort will be performed at the next time point for which ground control has previously supplied an abort solution. In such a case ground control will have the predicted CM landing point.

6. Block-data abort solutions for the SPS will be provided following each GO/NO GO decision point.

7. Block-data abort solutions will be used only in the event of complete communications loss.

8. Block data will be provided in a form and quantity such that the crew will have sufficient information to perform an SCS auto abort (to backup ISS or CMC failures).

   8a. The data will be compatible with outputs from the onboard return-to-earth program (P37).

   8b. The block-data solutions will involve inplane aborts only.

8c. Table VI is a schedule of block-data abort solutions, showing the time at which ground control provides the data, the approximate time of ignition, and the rationale for the solution.

8d. Table VII presents the information required for block data.

9. In the event that circumlunar, post-pericynthion aborts result in a shorter total flight time than direct-return aborts, the former will be used.

6.2 Procedures

In general terms, TLC abort situations can arise in two different ways: degradation of a life-support system at a rate too rapid to permit continuation of the mission, or loss of a spacecraft subsystem so as to preclude one or more mandatory mission objectives. Loss of ground/spacecraft communications constitutes a severe contingency of the second type, and procedures for the no-communications case will be presented in a separate logical flow diagram.

The logical flow charts presented in figure 6.2-1 and 6.2-2 are predicated on the assumption that no major trajectory dispersion or subsystem failure has been identified during translunar injection, the subsequent malfunction checks, or the transposition and docking (T and D) maneuvers. Immediately following T and D, the crew will have as a minimum an abort solution for TLI cutoff +5 hours, provided to cover SPS failures occurring during TLI and T and D. Between T and D and the first midcourse correction (MCC 1) this 5-hour solution will be updated to include TLI dispersions, and a 10-hour solution will be provided for earth return in case of an SPS failure at MCC 1. The schedule for providing and updating block-data solutions continues as indicated in table VII. Rationale for the solution schedule is as follows:

Return-to-earth solutions should be provided a few hours after each GO/NO GO decision time. These GO/NO GO points are:

1a. Immediately following MCC 1.

1b. Following the first two crew sleep periods.

1c. The point at which the spacecraft must be committed beyond the switchover time for direct versus circumlunar aborts using DPS ΔV capability. This latter decision point is variable, depending on earth/moon distance, i.e. launch date (see section 3.3.1).

2. Following each ΔV maneuver, solutions for later abort ignition times should be updated to incorporate the trajectory effects of the maneuver.

Table VI. SCHEDULE OF BLOCK DATA ABORT SOLUTIONS FOR TRANSLUNAR COAST

Time at Which Ground Control Supplies Data	Approximate Abort Ignition Time (Hrs after TLI cutoff)	Rationale for Providing Abort Solution at This Time
1. During earth parking orbit (Pre-TLI)	5	1. To provide data for SPS failures detected during TLI or T and D - CSM/LM configuration - abort ΔV within LM DPS capability including 200 fps reserve for MCC
2. Between T and D and MCC 1 (MCC at TLI cutoff + 7 hrs)	5	2a. Refine solution No. 1 by incorporation of TLI dispersions - DPS ΔV capability
	10	2b. To provide a return-to-earth solution in the event the SPS fails at MCC - solution includes no effects of a MCC ΔV
3. Following MCC 1	10	3a. Supply solution incorporating effects of MCC ΔV - DPS ΔV capability
	15	3b. Provide abort solution for SPS failure at 10-hr abort - for all later ignition times, a postpericynthion abort within the LM DPS capability returns faster than a direct-return abort
	20	3c. Solution provided at $\sim$2 hrs following crew sleep period - abort ΔV within SPS capability in CSM/LM docked configuration
	86	3d. A postpericynthion solution within DPS capability, providing backup for SPS failures occurring during remainder of TLC - t_{ig} is 2 hrs outside MSI on transearth leg of free-return trajectory (solution to be updated following subsequent MCCs)

Table VI. (Continued)

Time at Which Ground Control Supplies Data	Approximate Abort Ignition Time (Hrs after TLI cutoff)	Rationale for Providing Abort Solution at This Time
4. TLI cutoff + 19 hrs	44	4a. This solution provided ~2 hrs following crew sleep period - abort ΔV within SPS capability in CSM/LM docked configuration
	86	4b. Update to previous postpericynthion solution - within LM DPS capability
5. TLI cutoff + 43 hrs	86	5. Update to solution No. 4b. For t_{ig} later than TLI cutoff + about 50 hrs, direct-return aborts using SPS capability in docked configuration do not return as fast as postpericynthion (low ΔV) aborts
6. TLI cutoff + 61 hrs	86	6. Update solution No. 5 - still within LM DPS capability

Upon confirmation of an abort situation, the crew, with ground support, must establish a safe maximum return time and the ΔV required to achieve it. In the CSM/LM docked configuration the available ΔV for abort situations is approximately 5000 fps (SPS) or approximately 1900 fps (LM DPS). The SPS can provide 10,000 fps to the CSM only. This ΔV value determines whether the LM can be retained for backup capability. A minimum return time can be found for both the CSM and CSM/LM configurations, and the landing site control (if any) allowed by the maximum and minimum return times can be established. A real-time choice may have to be made between increased landing site control and the advantages of retaining the LM. For abort situations occurring between the direct/postpericynthion switchover time with DPS capability (TLI + 18 to 25 hrs) and the direct/postpericynthion switchover time with SPS capability (TLI + ~50 hrs), the SPS is the only means for decreasing return time below the postpericynthion value. Thus, during this interval, time-critical aborts will use the SPS.

6.2.1 Logical flow charts for ground communications case.- After the decision to abort has been made, and the return time/landing site tradeoffs have been made, ground control will send the abort solution via voice and computer uplink. In the voice-communication mode, the message would contain the information listed in table VIII which includes the external ΔV targeting parameters sent by telemetry uplink.

An abort solution via CMC uplink is accomplished through COLOSSUS program 27, which can be called from the ground or at astronaut option. Verification of receipt of the uplinked parameters is performed by calling P30 and subsequently P40 following an IMU alignment.

If the computer is inoperable, the abort maneuver will be performed in the SCS auto mode using the ΔV_c value supplied by ground control. The $\Delta \vec{V}$ alignment can be obtained by using the supplied gimbal angles, provided the IMU is found to be properly aligned. If not, $\Delta \vec{V}$ alignment may be established in one of two ways. If the IMU and one of the BMAG systems were known to be properly aligned within the previous three hours, the ground-supplied gimbal angles can be used in conjunction with the BMAG system to achieve the required alignment. Otherwise, $\Delta \vec{V}$ alignment must be obtained by use of the visual references for SPS ignition attitude. The logical flow diagram for abort cases with ground communication is shown in figure 6.2-1. A postabort MCC will be planned following all aborts from TLC. With good communications, MCC procedures will follow the procedures used for the abort maneuvers. Should communications be lost following the abort maneuver, any required MCC will follow the procedures outlined in section 6.2.2, with the exception that no block-data solution will be available.

6.2.2 Logical flow charts for no ground communications case.- If there is a failure of ground/spacecraft communications, the crew will perform an abort at the next time for which ground control has supplied a block-data solution. The block data will consist of the parameters shown in table VII. Program 37 will be called and an onboard solution will be generated using the block data solution (using the block data ΔV_c and GETI). Figure 6.2-2 is a detailed logical flow chart of the procedures to be followed for a TLC abort with no ground communications.

TABLE VII.- BLOCK DATA INFORMATION FOR TRANSLUNAR COAST

Parameter	Description
GETI	Ground elapsed time of abort maneuver ignition, hr:min:sec
T_{ig}	Impulsive ignition time for P37 input
ΔV_c	Change in velocity magnitude, fps
ϕ_L	Latitude of resultant landing point, + north, deg
λ_L	Longitude of resultant landing pointing, + east, deg
T_{FF}	Transit time from GETI to 400 000-ft altitude, hr:min:sec
V_{EI}	Inertial velocity at 400 000-ft altitude, fps
γ_{EI}	Inertial flight path angle at 400 000-ft altitude, deg
P, Y, R	IMU gimbal angles at SPS ignition attitude based on current REFSMMAT, deg
STAR/S	Identification of known STAR/S at ignition attitude to be used for BMAG alignment for SCS auto abort maneuver (for IMU fail) and IMU alignment verification
SHAFT TRUNNION	SHAFT and TRUNNION angles for BMAG alignment (for IMU fail) and IMU alignment verification

TABLE VIII.- RETURN TO EARTH MANEUVER PAD MESSAGE

Parameter	Description
Propulsion system	Propulsion system to be used
Guidance	Normally external ΔV
GETI	Ground elapsed time of main engine ignition
ΔV_x, ΔV_y, ΔV_z	Three components of impulsive ΔV along local vertical axis
ΔV_T	Total characteristic velocity change for the maneuver including ullage and tailoff
ΔT_B	Main engine on time
V_C	Characteristic velocity measured along X_b
R, P, Y	IMU gimbal angles at burn attitude
h_a, h_p	Post abort apogee and perigee altitudes
SXTS	Star identification for sextant for backup attitude and IMU check
SFT	Shaft angle of SXTS
TRN	Trunnion angle of SXTS
GET, LAT, LONG, ALT	Time and position for preburn state vector check
WGT, PTRM, YTRM	Vehicle weight engine trim angles

TABLE VIII.- RETURN TO EARTH MANEUVER PAD MESSAGE - Concluded

In addition to the maneuver pad, preburn entry data will be provided:

Parameter	Description
Area	
V_{400K}, γ_{400K}	CLA identification inertial entry vector
L.V.	Lift vector orientation at 400 000-ft altitude
R_{400K}, P_{400K}, Y_{400K}	IMU gimbal angles at 400 000-ft altitude
R_{tgo}.05g	Range to go to landing target at .05g
V_{IO}.05g	Inertial velocity at .05g
RET .05g	Time from beginning of main engine on to .05g
LAT, LONG	Geographic position of landing target
V_L	Skip velocity-inertial velocity at the point where g level falls below 0.2g
CONST G	Entry g level for backup entry
MAX G	Maximum load factor during entry
RETBBO	Time from GETI to beginning of blackout
RETEBO	Time from GETI to end of blackout
RETDROG	Time from GETI to drogue chute deployment
RETGI	Time from GETI to entry guidance initiation

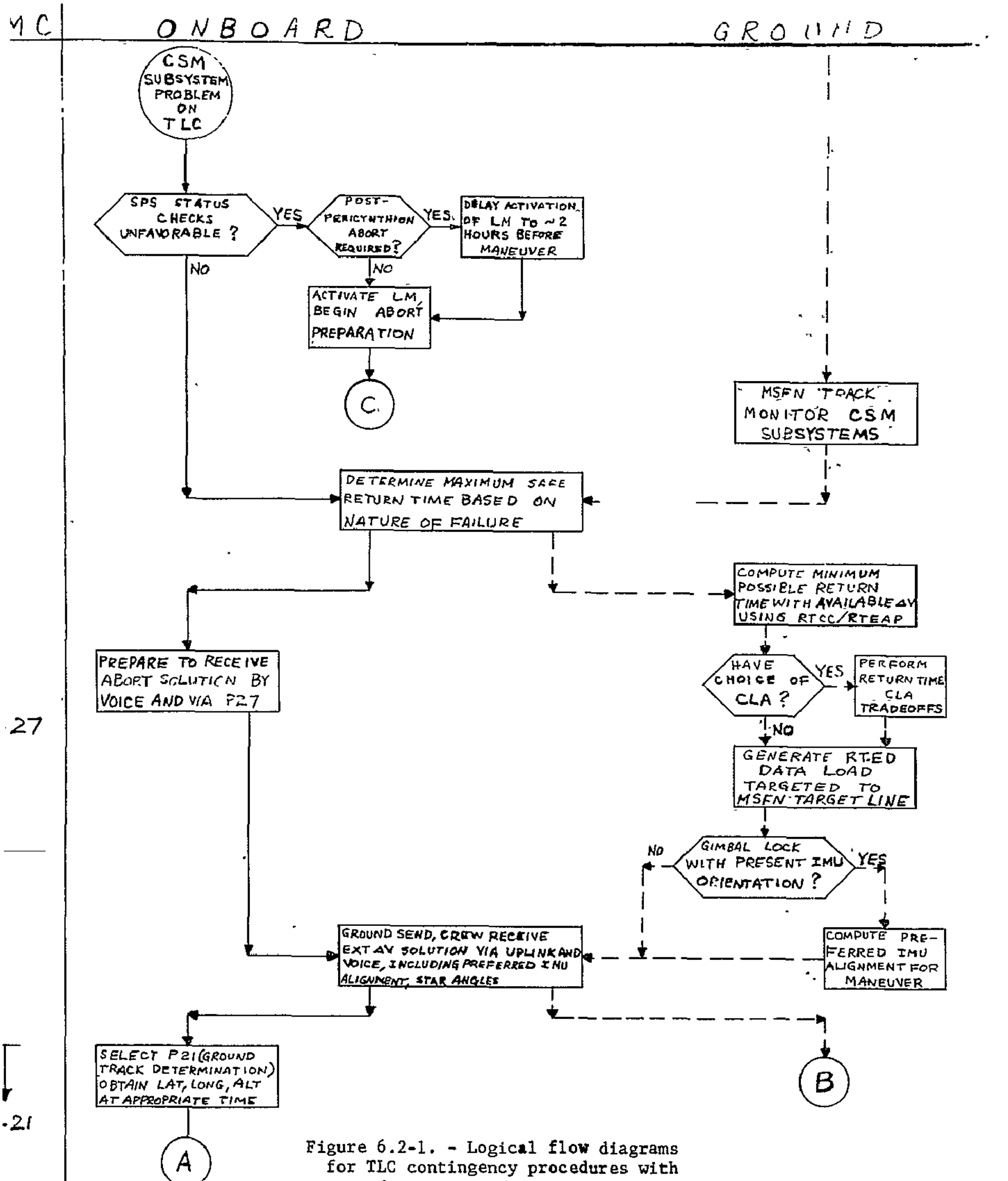


Figure 6.2-1. - Logical flow diagrams for TLC contingency procedures with ground to spacecraft communications.

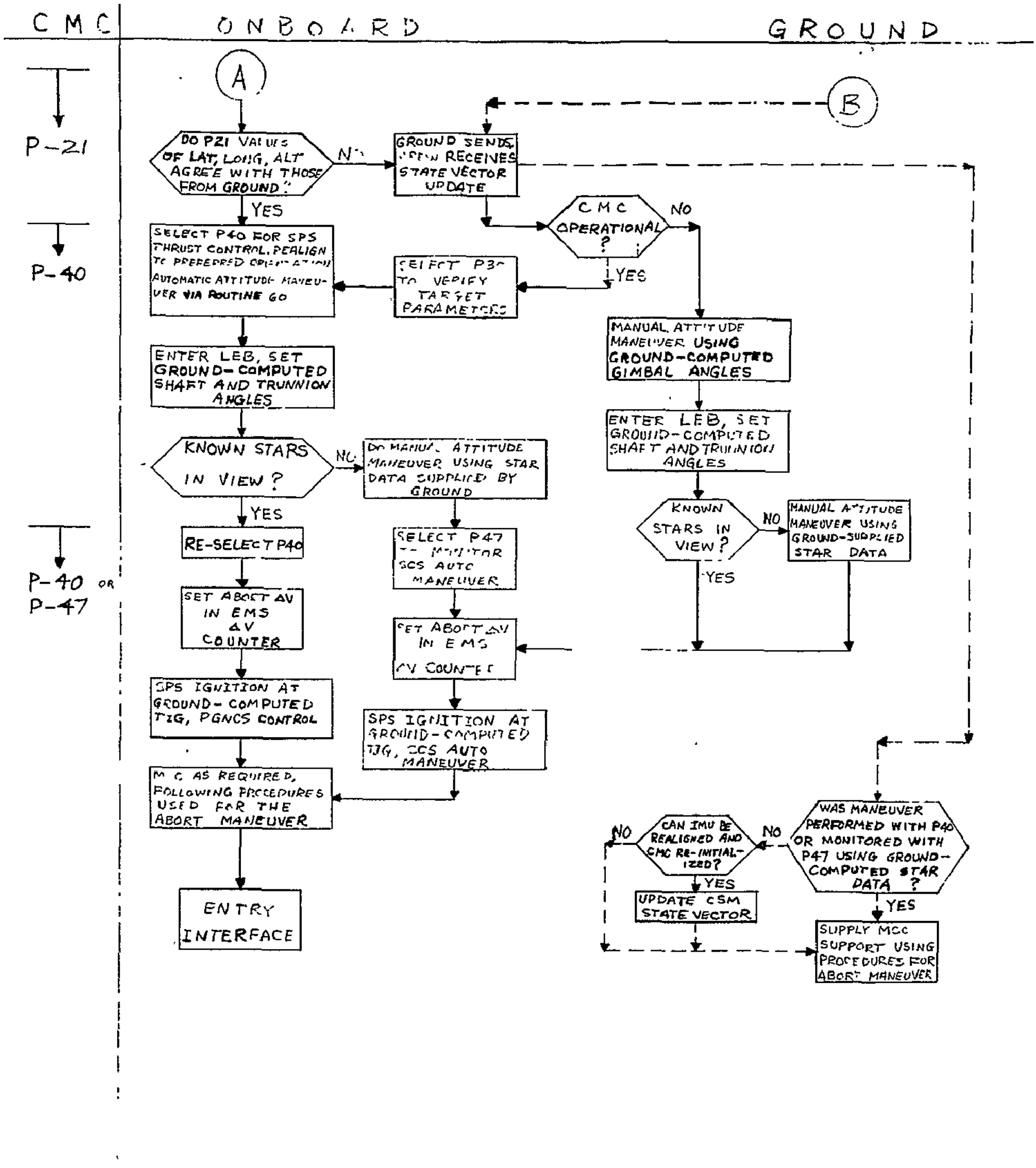


Figure 6.2-1.- Continued

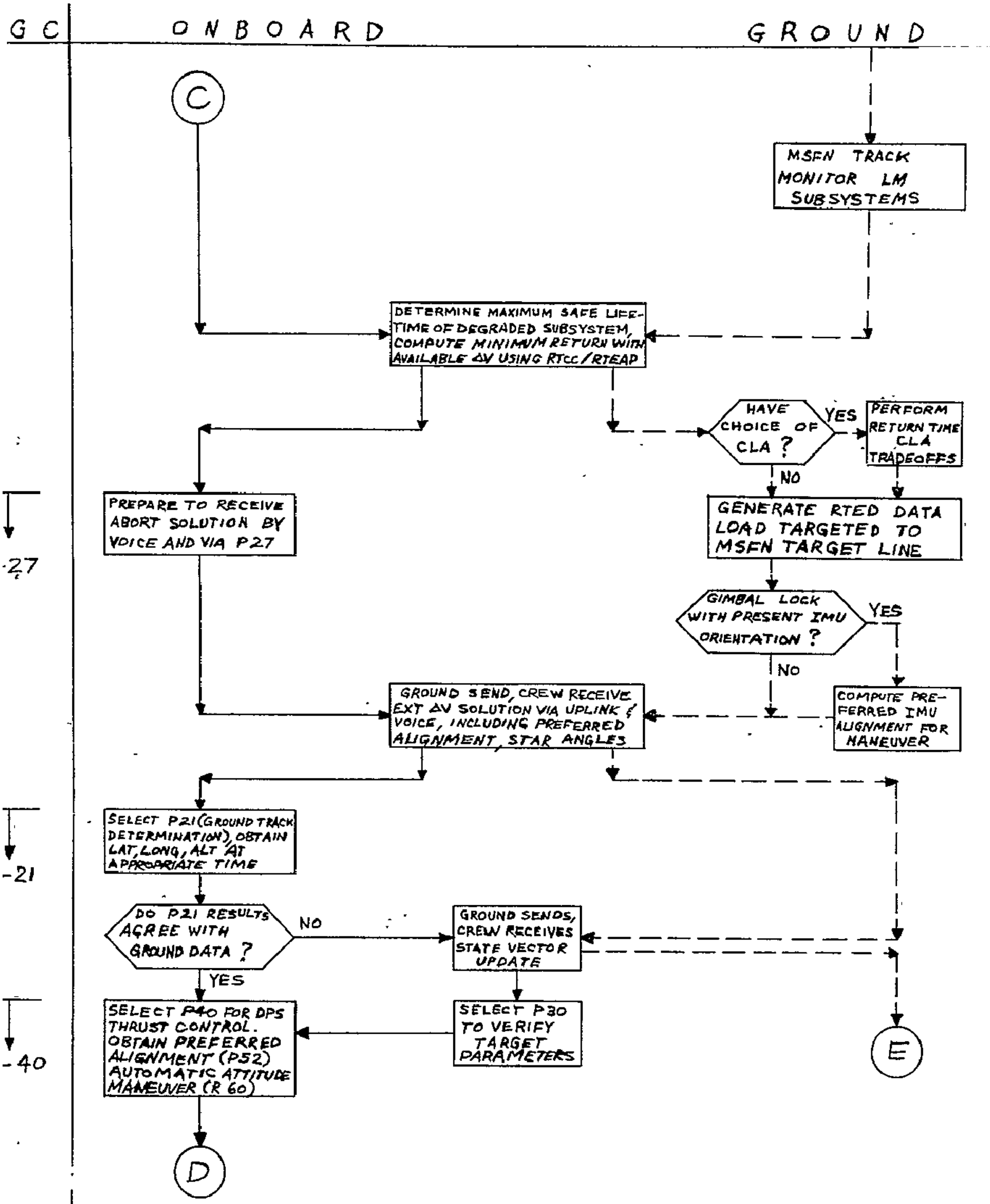


Figure 6.2-1.- Continued

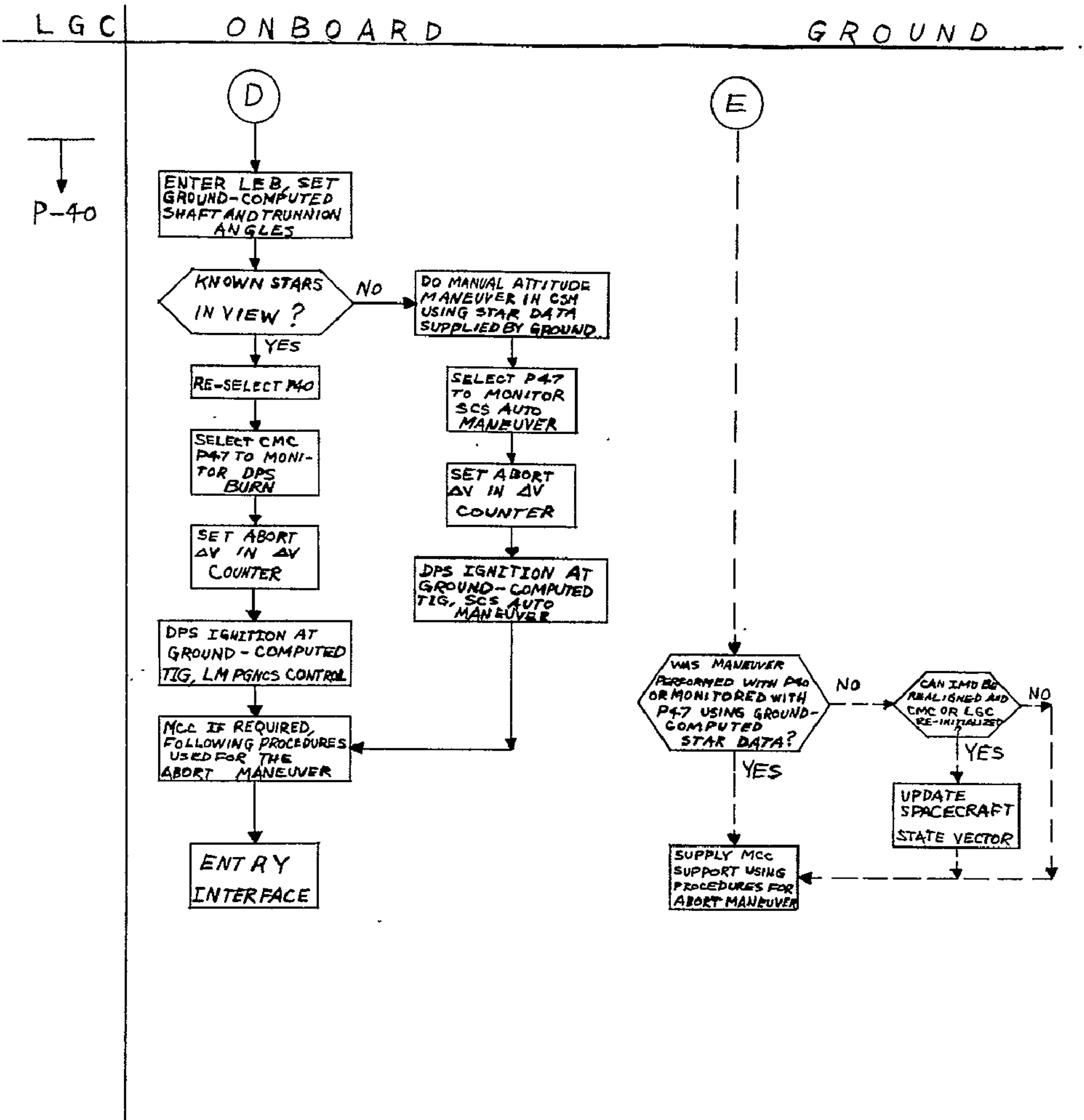


Figure 6.2-1.- Concluded

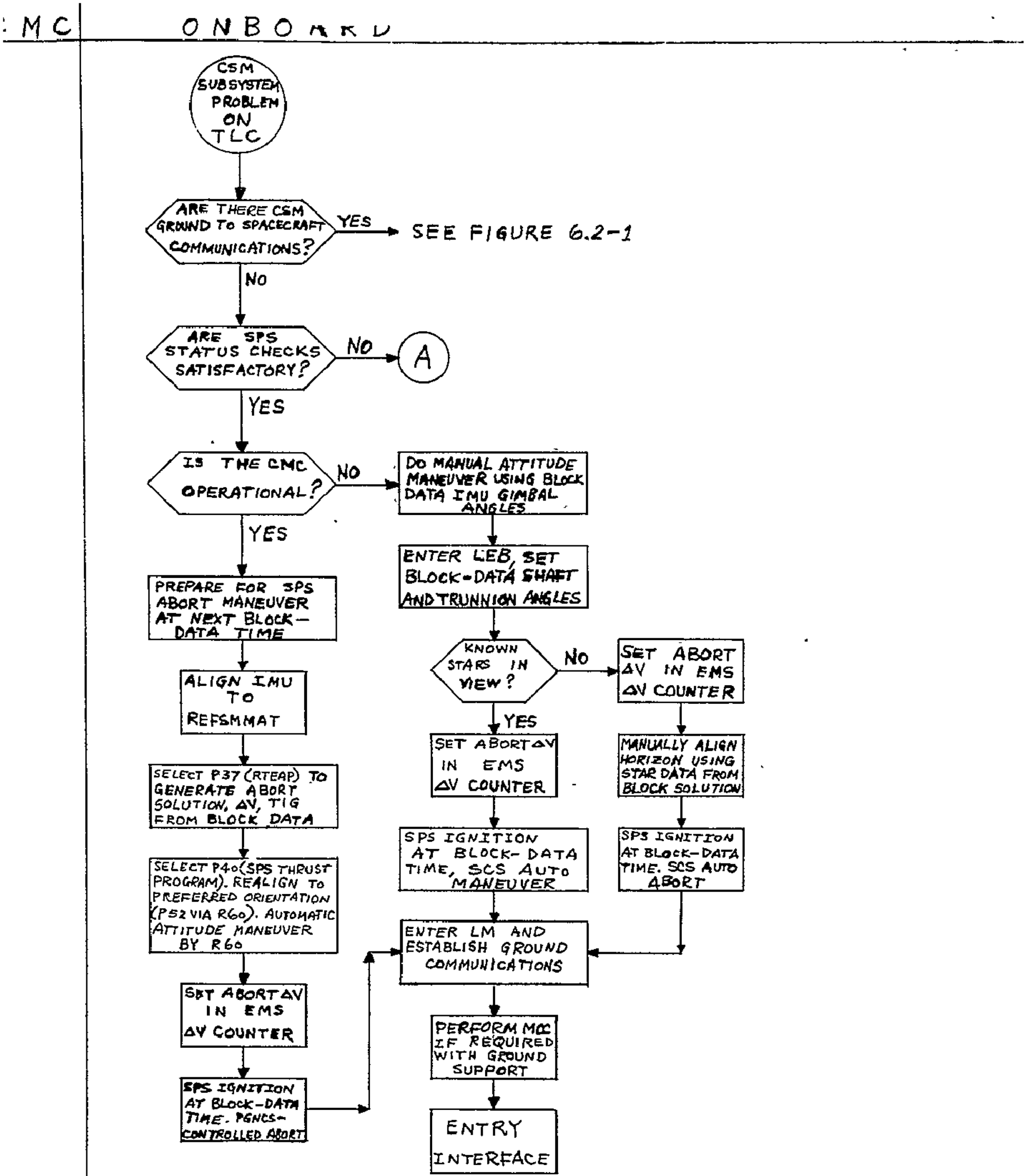


Figure 6.2-2.- Logical flow diagrams for TLC contingency procedures with no ground to spacecraft communications.

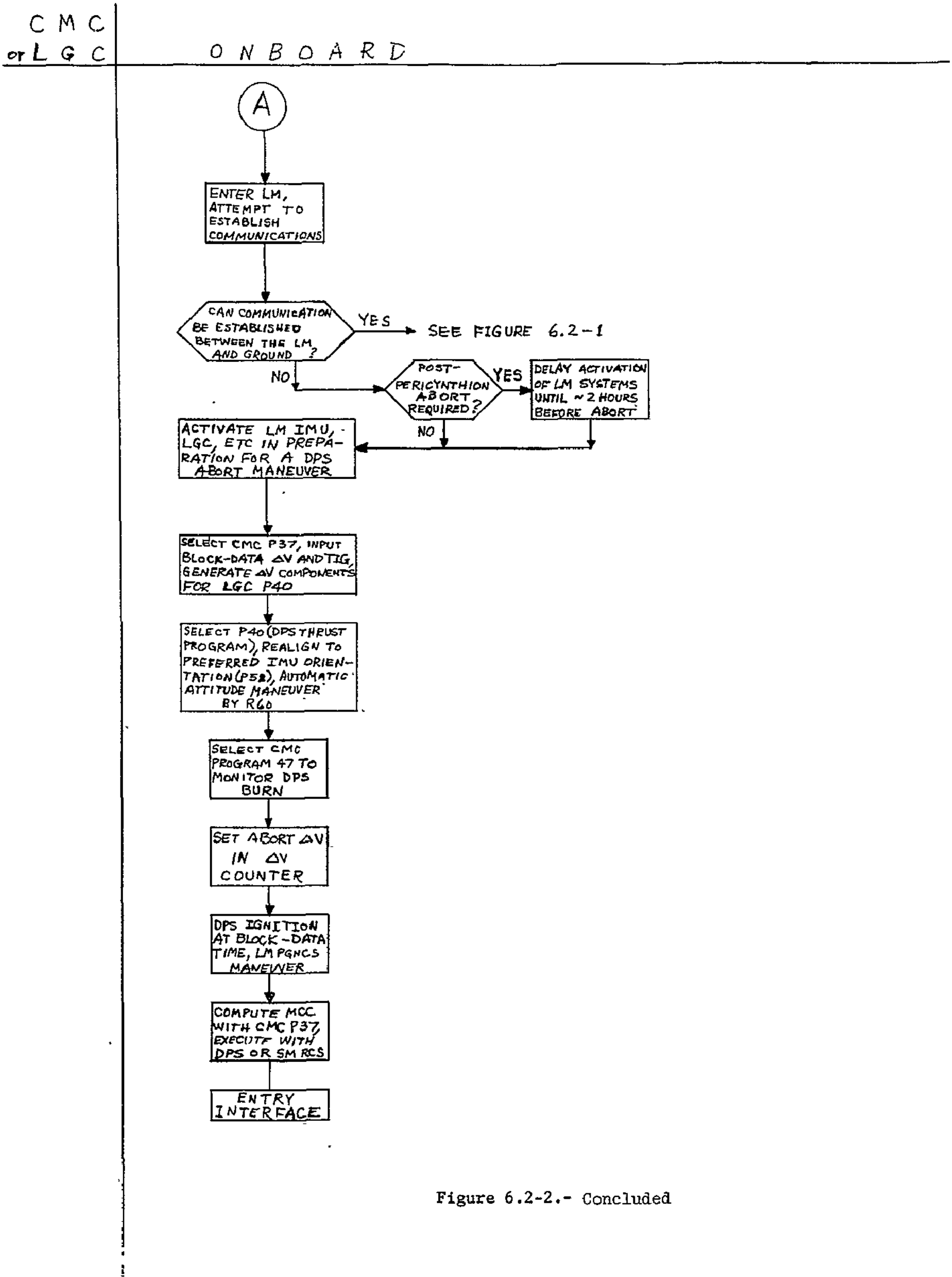


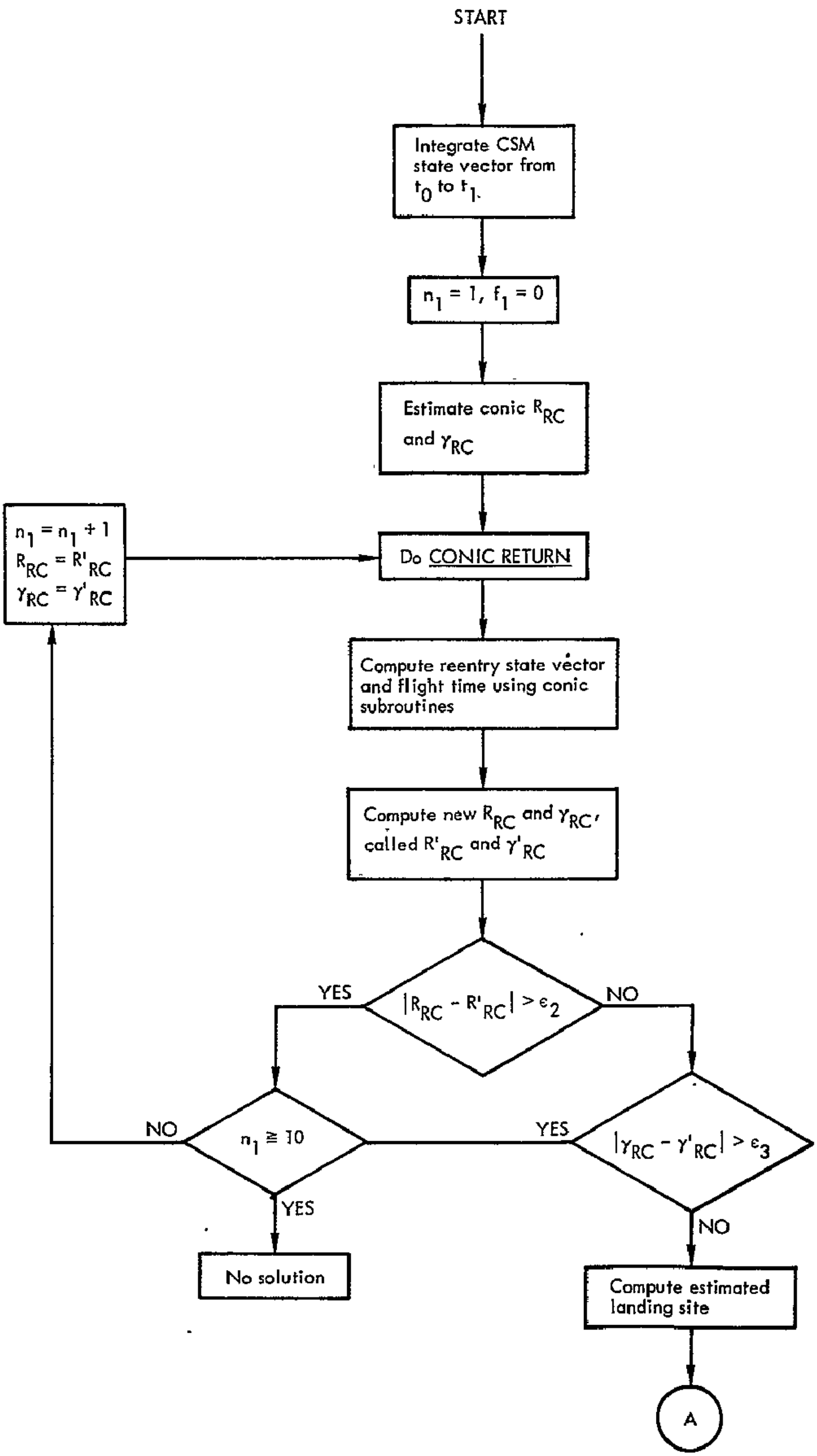
Figure 6.2-2.- Concluded

APPENDIX

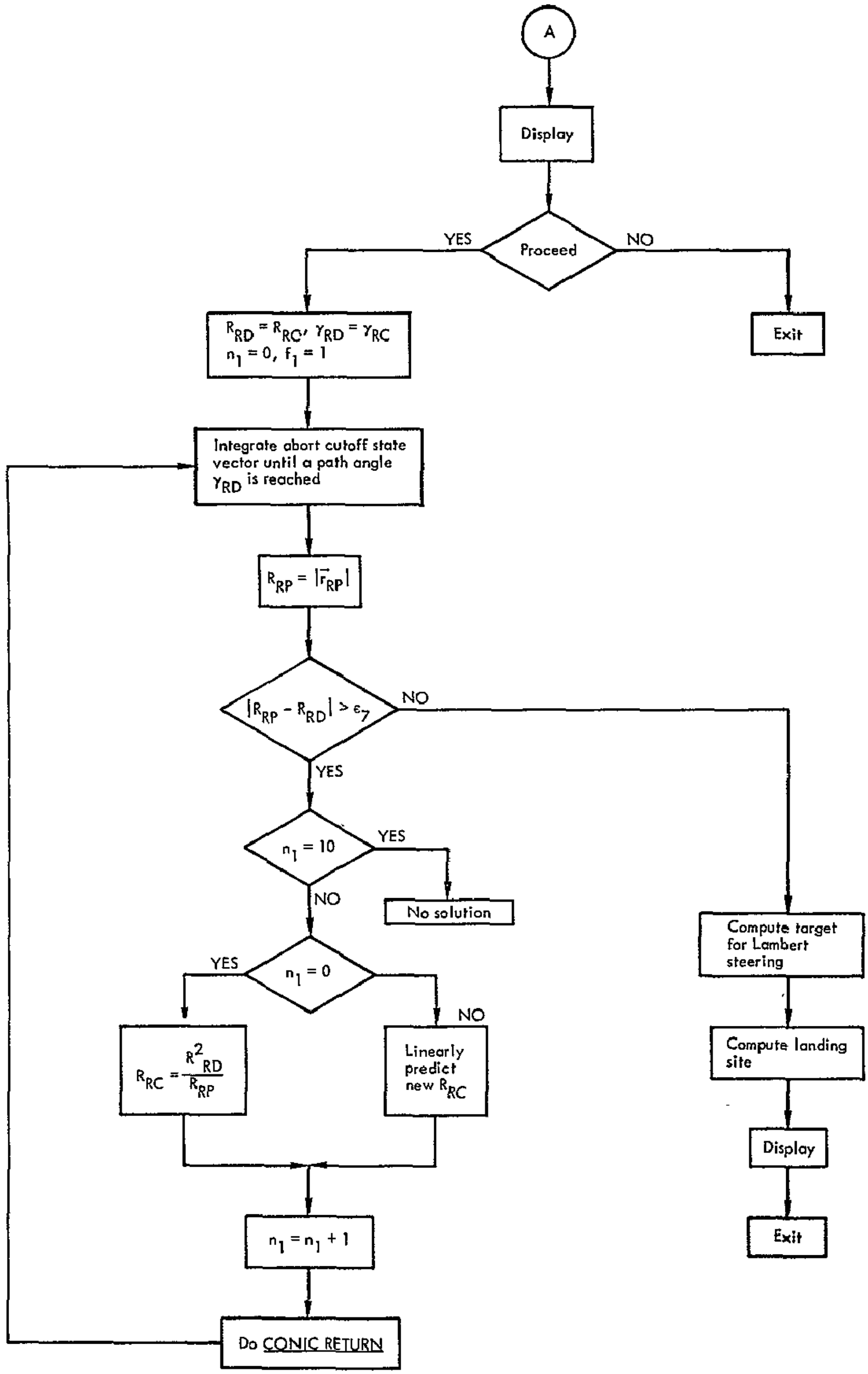
Figure A-1 shows the simplified logic diagram for the COLOSSUS return-to-earth targeting routine (P37), taken from reference 12. Figure A-1 is accompanied by a nomenclature list. The remainder of the Appendix is a listing of the operational interface between the CMC logic for P37 and the astronaut check list. This listing is reprinted from reference 13.

NOMENCLATURE for Figure A-1

t_o	Any CSM state-vector time during non-thrusting flight, as recorded by the CMC clock
t_1	Abort ignition time
n_1	Counter
f_1	Flag set to $\begin{cases} 0 \text{ initial conic phase} \\ 1 \text{ precision trajectory phase} \end{cases}$
R_{RC}	Conic entry radius magnitude
γ_{RC}	Conic entry flight path angle
R_{RD}	Desired final entry radius magnitude
γ_{RD}	Desired final entry flight path angle
R_{RP}	Final entry radius magnitude of a precision trajectory
$\vec{r}_{RP}$	Entry radius vector of a precision trajectory
$X(t_1)$	Cotangent of the flight path angle at abort cutoff
ΔV	Impulsive velocity change
$\vec{V}$	Velocity vector
ΔV_D	Desired velocity change

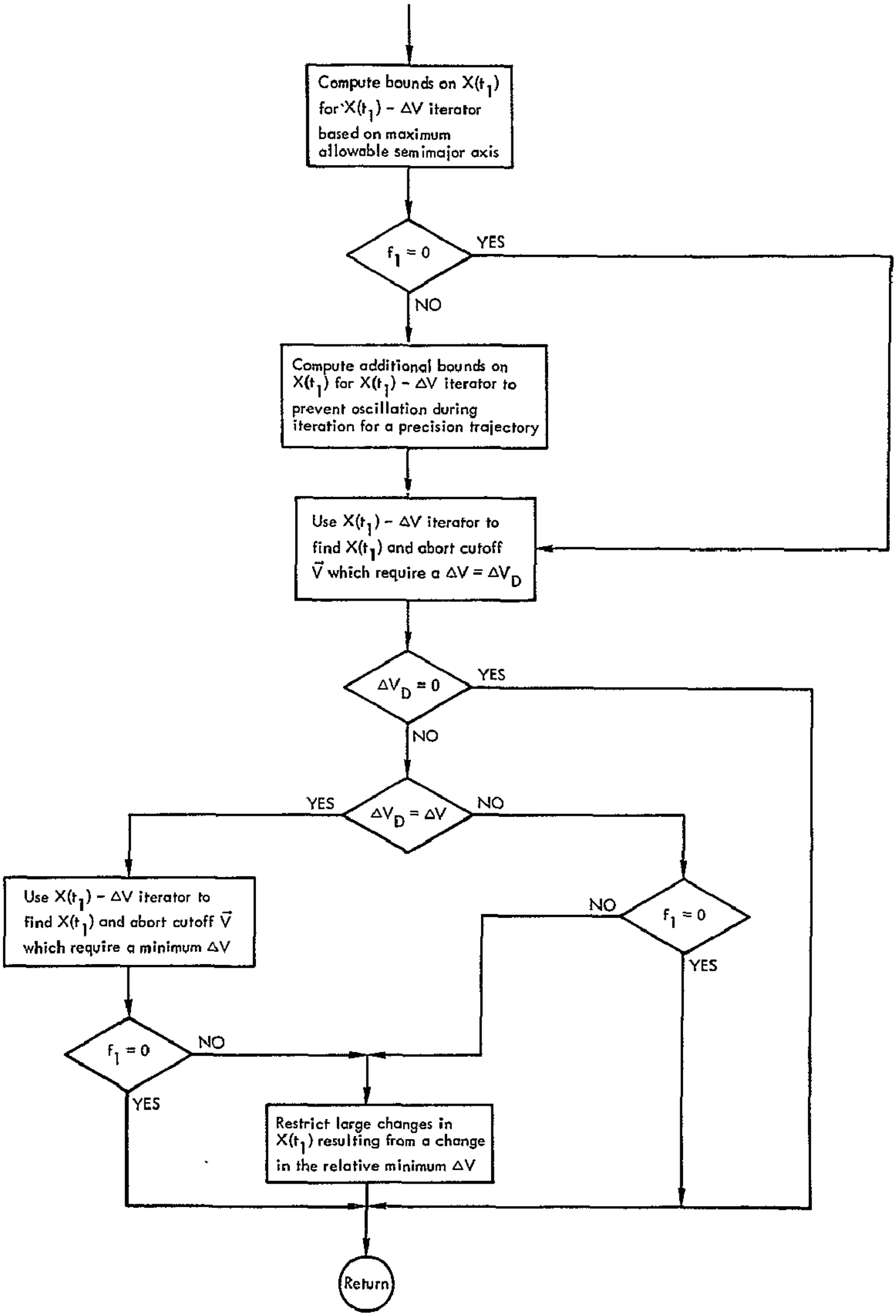


(Reference 12)



(Reference 12)

Figure A-1. - Continued



(Reference 11)

CONIC RETURN ROUTINE

Figure A-1. - Concluded

RETURN TO EARTH (P37)

LOGIC REV 20 04/09/68
CHECKLIST REV 09 12/27/67

PURPOSE: (1) THIS PROGRAM WILL COMPUTE A RETURN TO EARTH TRAJECTORY PROVIDING THE CSM IS OUTSIDE THE LUNAR SPHERE OF INFLUENCE AT THE TIME OF IGNITION.

(2) THIS PROGRAM COMPUTES AND DISPLAYS A PRELIMINARY SERIES OF PARAMETERS BASED ON A CONIC TRAJECTORY AND:

A. ASTRONAUT SPECIFIED TIME OF IGNITION.

B. ASTRONAUT SPECIFIED MAXIMUM CHANGE IN VELOCITY.

C. ASTRONAUT SPECIFIED RE-ENTRY ANGLE.

THESE PARAMETERS ARE:

(A) TIME FROM IGNITION TO REENTRY.

(B) REENTRY INERTIAL VELOCITY.

(C) REENTRY FLIGHT PATH ANGLE.

(D) LATITUDE OF SPLASH.

(E) LONGITUDE OF SPLASH.

(F) DELTA V (LV)

(3) WHEN THE INITIAL DISPLAY IS SATISFACTORY TO THE ASTRONAUT, THE PROGRAM RECOMPUTES THE SAME DATA USING APPLICABLE PERTURBATIONS TO THE CONIC TRAJECTORY, AND DISPLAYS THE NEW VALUES.

(4) UPON FINAL ACCEPTANCE BY THE ASTRONAUT, THE PROGRAM COMPUTES AND STORES THE TARGET PARAMETERS FOR RETURN TO EARTH FOR USE BY SPS PROGRAM (P40) OR RCS PROGRAM (P41).

(5) BASED UPON SPECIFIED PROPULSION SYSTEM THE FOLLOWING ARE DISPLAYED:

(A) MIDDLE GIMBAL ANGLE AT IGNITION.

(B) TIME OF IGNITION (TIG).

(C) TIME FROM IGNITION (TFI).

ASSUMPTIONS: (1) THIS PROGRAM ASSUMES THAT CONTACT WITH THE GROUND IS UNAVAILABLE, AND IS COMPLETELY SELF CONTAINED.

(2) THE ISS NEED NOT BE ON TO COMPLETE THIS PROGRAM.

++
+20 (3) IF VALUE OF VPRED ENTERED IN NOUN 60 IS LESS THAN THE MINIMUM REQUIRED TO RETURN TO EARTH, THE DELTA V REQUIRED VECTOR WILL BE COMPUTED BASED ON A MINIMUM VALUE. IF VALUE ENTERED IS GREATER THAN THE MINIMUM REQUIRED TO RETURN TO
\+ EARTH THEN THE ASTRONAUT DESIRED VALUE WILL BE USED TO COMPUTE THE DELTA V REQUIRED VECTOR. THE COMPUTED DELTA V
+20 REQUIRED VECTOR WILL BE DISPLAYED IN NOUN 81.
++

(4) THE DAP DATA LOAD ROUTINE SHOULD BE PERFORMED PRIOR TO COMPLETION OF THIS PROGRAM (SEE CREW COLUMN BELOW).

(5) THIS PROGRAM IS SELECTED BY THE ASTRONAUT BY DSKY ENTRY.

PROG CONT	CMC	GROUND	CREW	CHECKLIST	TIME	TOTAL TIME
			CREW PROG. SELECTION			
						#10
	START RETURN TO EARTH PROGRAM (P37) DISPLAY PROGRAM 37		KEY IN RETURN TO EARTH PROGRAM (P37) V37E 37E			
						#20
			MONITOR DSKY: OBSERVE DISPLAY OF PROGRAM 37			
++ +19 ++	SET ECSTEER = 1.					#30
	"A"		"A"			
						#40
HOLD SNAP	FLASH VERB-NOUN TO REQUEST RESPONSE AND DISPLAY TIG: V06 N33 R1-TIG-HRS R2-TIG-MIN R3-TIG-SEC		MONITOR DSKY: OBSERVE VERB-NOUN FLASH TO REQUEST RESPONSE AND DISPLAY OF TIG.			#50

TIG-TIME OF
IGNITION (GET).
IN HRS. MIN. SEC
TO NEAREST .01 SEC

AM I SATISFIED WITH
THIS VALUE?

Y

N

RECORD THIS
VALUE

WAIT FOR KEYBOARD
ENTRY

KEY IN PROCEED

TERMINATE FLASH UPON
RECEIPT OF PROCEED
OR NEW DATA

KEY IN V25E AND
LOAD THE DESIRED
TIG.

P.
R.
O.
C.
E.
E.
D.

NEW
DATA

STORE DATA

HOLD .
SNAP .

FLASH VERB-NOUN TO
REQUEST RESP
DISPLAY VPRE
GAMMA EI:
V06 N60
R1-BLANK
R2-VPRED
R3-GAMMA

MONITOR DSKY:
OBSERVE VERB-NOUN
FLASH TO REQUEST
RESPONSE AND DISPLAY
OF VPRED AND GAMMA
EI

VPRED-MAXIMUM ALLOW-
ABLE CHANGE IN
VELOCITY IN FPS TO
NEAREST FPS

GAMMI EI-FLIGHT PATH
ANGLE BETWEEN INER-
TIAL VELOCITY VECTOR
AND THE LOCAL HOR-
IZONTAL AT THE ENTRY
INTERFACE ALTITUDE
AT 400,000 FT. IN DE-
GREES TO THE NEAR-
EST .01 DEG. MINUS
INDICATES FLIGHT
PATH IS BELOW THE
HORIZONTAL PLANE.
NOTE: IF ZERO IS
LOADED COMPUTATION
WILL BE SOLVED TO
HIT CENTER OF ENTRY
CORRIDOR, OTHERWISE
COMPUTATION WILL BE
SOLVED TO HIT ANGLE
ENTERED.

AM I SATISFIED WITH
THESE VALUES?
NOTE: IN ORDER TO
HAVE THE CMC COMPUTE
A MINIMUM ENERGY
(MINIMUM FUEL) RE-
TURN TO EARTH MAN-
EUVER THE ASTRONAUT
SHOULD LOAD ALL
ZEROS INTO R2.
IN ORDER TO HAVE THE
CMC COMPUTE A TRAJ-
ECTORY WHICH WILL
HIT THE CENTER OF
THE ENTRY CORRIDOR
THE ASTRONAUT SHOULD
LOAD ALL ZERO'S INTO
R3.

Y

N

KEY IN PROCEED

WAIT FOR KEYBOARD
ENTRY

```
                                    .                                   .   .
                                    .                                   .   .
                                    .                     ----------------- .
          TERMINATE FLASH UPON      .          .          KEY IN V22E, V23E,  .
          RECEIPT OF PROCEED        .     ...........     OR V25E AND LOAD    .
          OR NEW DATA               .          .          THE DESIRED VALUES. .
          -------------------       .                     ----------------- .
           .P           .NEW        .                               .       .
           .R           .DATA       .                               .       .
           .O           .           .                               .       .
           .C           .           .                               .........
           .E      ----------       .
           .E      STORE DATA       .
           .D      ----------       .
           .            .           .
           .            .           .
           .            .           .
           .            .............
           .
           .
           .
          -------------------
          BASED ON ASTRONAUT
          INPUTS USE CONIC
          SECTION METHOD TO
          COMPUTE NECESSARY
          DEPENDENT VARIABLES
          FOR EVALUATION OF
          THRUSTING MANEUVER
          INCLUDING ENTRY LAT,
          LONG, FLIGHT PATH
          ANGLE, VELOCITY,
          AND TIME TO ENTRY.
             ESTABLISH ALARM
++        IF-
+20          (A) STATE VECTOR
+         AT TIG IS IN MOON'S
+         SPHERE OF INFLUENCE.
+         STORE ALARM CODE 612.
+
+            (B) SOLUTION DOES
+         NOT CONVERGE DUE TO
+         EXCESSIVE ITERATIONS.
+         STORE ALARM CODE 605.
+
+            (C) DESIRED
+         FLIGHT PATH ANGLE
+20       NOT REACHED.
++        STORE ALARM CODE 613
          -------------------
          .N     .A
          .O     .L
          .      .A
          .A     .R
          .L     .M
          .A     .
          .R     .
          .M     .
          .      .
          .      .
          .      .
          .      .
```

```
        .        .
        .        .
        .        .
        .        .     "B"
        .        .      .
        .        .      .
        .        .      .
        .        .     ...
        .        .      .
        .        . -----------
        .        . RECOMPUTE
        .        . NECESSARY
        .        . DEPENDENT
        .        . VARIABLES
        .        . FOR PERFORM-
        .        . ING THRUST-
        .        . ING MANEUVER
        .        . USING PRECI-
        .        . SION INTEG-
        .        . RATION
        .        .   ESTABLISH
        .        . ALARM IF A
        .        . SOLUTION
        .        . DOES NOT
        .        . CONVERGE DUE
        .        . TO EXCESSIVE
        .        . ITERATIONS.
        .        . -----------
        .        .    .A      .N
        .        .    .L      .O
        .        .    .A      .
        .        .    .R      .A
        .        .    .M      .L
        .        .    .       .A
        .        .    .       .R
        .        .    .       .M
        .        .    .       .
        .        .    .       .
        .        .    .       .                           "B"
        .        .    .       .                            .
        .        .    .       .                            .
        .        .    .       .                            .
        .        .    .       .                           ...
        .        .    .       .                            .
        .  --------------     .                   ------------------
.HOLD . .  FLASH VERB-        .          .        MONITOR DSKY:
....... .  NOUN TO RE-        .    ...........       IS THERE AN ALARM
 SNAP . .  QUEST RESPONSE     .          .        CODE DISPLAY INDICA-
        .  AND DISPLAY        .                   TING COMPUTATIONAL
        .  ALARM CODE         .                   DIFFICULTY?
        .    V05 N09          .                   ------------------
        .    R1-              .                    .Y          .N
        .    R2-              .                    .           .
        .    R3-              .                    .           .
        .                     .                    .           .
        .                     .                    .           .
        .                     .                    .           .
        .                     .                    .           .
        .                     .                    .           .
        .                     .                    .           .
        .                     .                    .           .
```

++
+20
++

EXPECTED ALARM CODE AT THIS TIME IS AS DEFINED ABOVE.

WAIT FOR KEY-BOARD ENTRY

TERMINATE FLASH UPON RECEIPT OF RECYCLE

RECYCLE

GO TO "A" ABOVE

RETURN TO START OF PROGRAM AND ADJUST INPUT PARAMETERS. KEY IN RECYCLE V32E

RECYCLE

GO TO "A" ABOVE

HOLD .
....
SNAP .

FLASH VERB-NOUN TO REQUEST RESPONSE AND DISPLAY COMPUTED DATA
V06 N61
R1-IMPACT LAT
R2-IMPACT LONG
R3-BLANK

MONITOR DSKY: OBSERVE VERB-NOUN FLASH TO REQUEST RESPONSE AND DISPLAY OF S/C ENTRY DATA

IMPACT LAT-LATITJDE OF CALCULATED IMPACT POINT. IN DEGREES TO NEAREST .01 DEG. + IS NORTH

IMPACT LONG-LONGI-TUDE OF CALCULATED IMPACT POINT. IN DEGREES TO NEAREST .01 DEG. + IS EAST

AM I SATISFIED WITH THESE VALUES?

Y

N

RECORD THESE VALUES.

WAIT FOR KEYBOARD ENTRY

KEY IN PROCEED

RETURN TO START OF PROGRAM AND ADJUST INPUT PARAMETERS KEY IN RECYCLE V32E

TERMINATE FLASH UPON RECEIPT OF PROCEED OR RECYCLE

PROCEED

RECYCLE

GO TO "A" ABOVE

GO TO "A" ABOVE

```
                .
                .
                .
        ------------------                  ------------------
        FLASH VERB-NOUN TO                  MONITOR DSKY:
        REQUEST RESPONSE AND              . OBSERVE VERB-NOUN
HOLD .  DISPLAY DELTA T     ................. FLASH TO REQUEST
....... TRANS:                            . RESPONSE AND DISPLAY
SNAP .     V06 N39                          OF DELTA T TRANS
           R1-DELTA T TRANS-                ------------------
           HRS                                      .
           R2-DELTA T TRANS-                        .
           MINS                                     .
           R3-DELTA T TRANS-                        .
           SECS                             ------------------
                                            AM I SATISFIED
        DELTA T TRANS - TIME                WITH THIS VALUE?
        REQUIRED FOR TRANS-                 ------------------
        FER FROM TIG TO TIME                        .Y      .N
        OF REENTRY AT 400,000                       .       .
        FT. ALT. (REFERENCED                        .       .
        TO FISCHER ELLIPSOID)                       .       .
        IN HRS, MINS, SEC TO                ---------------  .
        NEAREST .01 SEC.                    RECORD THIS      .
        ------------------                  VALUE.           .
                .                           ---------------  .
                .                                   .        .
                .                                   .        .
                .                                   .        .
        ------------------                  ---------------  .
        WAIT FOR KEYBOARD   ................. KEY IN PROCEED   .
        ENTRY                 .             ---------------  .
                                                             .
                                                             .
                                                             .
                                              ------------------
                                              RETURN TO START
                                              OF THIS PROGRAM
                              .               AND ADJUST INPUT
        TERMINATE FLASH UPON .................. PARAMETERS KEY IN
        RECEIPT OF PROCEED    .               RECYCLE
        OR RECYCLE                            V32E
        ------------------                    ------------------
        .P      .R                                    .
        .R      .E                                    .
        .O      .C                                    .
        .C      .Y                                   ...
        .E      .C                                    .
        .E      .L                                  GO TO
        .D      .E                                   "A"
        .        .                                  ABOVE
        .        .
        .       ...
        .        .
        .      GO TO
        .       "A"
        .      ABOVE
        .
        .
        .
        .
        .
        .
```

HOLD
SNAP
++
+20
++

FLASH VERB-NOUN TO REQUEST RESPONSE AND DISPLAY VPRED AND GAMMA EI:
V06 N60
R1-BLANK
R2-VPRED
R3-GAMMA EI

VPRED - PREDICTED INERTIAL VELOCITY AT THE ENTRY INTERFACE (400,000 FT ABOVE THE FISCHER ELLIPSOID) IN FPS TO THE NEAREST FPS

GAMMA EI - SEE ABOVE DISPLAY FOR DEFINITION

MONITOR DSKY: OBSERVE FLASH TO REQUEST RESPONSE AND DISPLAY OF ENTRY PARAMETERS

AM I SATISFIED WITH THESE VALUES?

Y

N

RETURN TO START OF THIS PROGRAM AND ADJUST INPUT PARAMETERS KEY IN RECYCLE V32E

WAIT FOR KEYBOARD ENTRY

RECORD THESE VALUES.

GO TO "A" ABOVE

TERMINATE FLASH UPON
RECEIPT OF PROCEED
OR RECYCLE

KEY IN PROCEED

PROCEED

RECYCLE

GO TO
"A"
ABOVE

HOLD
SNAP

FLASH VERB-NOUN TO
REQUEST RESPONSE AND
DISPLAY THREE STORED
COMPONENTS OF
DELTA V(LV):
V06 N81
R1-DELTA VX(LV)
R2-DELTA VY(LV)
R3-DELTA VZ(LV)

MONITOR DSKY:
OBSERVE VERB-NOUN
FLASH TO REQUEST RE-
SPONSE AND DISPLAY
OF THREE STORED
COMPONENTS OF IMPUL-
SIVE DELTA V ALONG
CSM LOCAL VERTICAL
AXES AT TIG

DELTA VX(LV)-COMPO-
NENT OF IMPULSIVE
DELTA V AT TIG ALONG
(RXV)XR. IN FPS TO
NEAREST .1 FPS

DELTA VY(LV)-COMPO-
NENT OF IMPULSIVE
DELTA V AT TIG ALONG
VXR. IN FPS TO NEAR-
EST .1 FPS

RECORD THESE VALUES.

DELTA VZ(LV)-COMPO-
NENT OF IMPULSIVE
DELTA V AT TIG ALONG
-R. IN FPS TO NEAR-
EST .1 FPS

```
WHERE R IS CSM
GEOCENTRIC RADIUS
VECTOR AND V IS CSM
INERTIAL VELOCITY
VECTOR AT TIG.
-------------------

-------------------                    -------------
WAIT FOR KEYBOARD  ................... KEY IN PROCEED
ENTRY                                  -------------

TERMINATE FLASH UPON
RECEIPT OF PROCEED
-------------------
 .P
 .R
 .O
 .C
 .E
 .E
 .D

-------------------                    -------------------
WAS THIS THE FIRST                     WAS THIS THE FIRST
PASS THROUGH THE                       PASS THROUGH THE
PROGRAM?                               PROGRAM?
-------------------                    -------------------
  .N          .Y                         .Y     .N

           GO TO                       GO TO
            "B"                         "B"
           ABOVE                       ABOVE

--------------
SET CMC ASSUMED
OPTION IN R2
(BELOW) TO
00001
--------------
```

FLASH VERB-NOUN TO REQUEST RESPONSE AND DISPLAY OPTION CODE FOR PROPULSION SYSTEM OPTION:
V04 N06
R1-00007
R2-0000X
R3-BLANK

HOLD
SNAP

R1-IS THE OPTION CODE FOR ASSUMED PROPULSION SYSTEM OPTION

R2- IS CMC ASSUMED OPTION

00001-SPS PROPULSION SYSTEM

00002-RCS PROPULSION SYSTEM

MONITOR DSKY:
OBSERVE VERB-NOUN FLASH TO REQUEST RESPONSE AND DISPLAY OF OPTION CODE FOR PROPULSION SYSTEM OPTION

AM I SATISFIED WITH THE CMC ASSUMED OPTION?

Y

N

PRIOR TO PROCEEDING ON THIS DISPLAY THE DAP DATA LOAD ROUTINE (R03) SHOULD HAVE BEEN COMPLETED IN ORDER THAT THIS PROGRAM USE THE CORRECT VALUES FOR WEIGHT AND, IN THE CASE OF RCS, NUMBER OF JETS TO BE USED DURING COMPUTATION OF TIME OF BURN.

WAIT FOR KEYBOARD ENTRY

KEY IN PROCEED

TERMINATE FLASH UPON
RECEIPT OF PROCEED
OR NEW DATA

PROCEED

NEW DATA

STORE
DATA

KEY IN V22E
AND LOAD THE
DESIRED OP-
TION

BASED ON THRUST
OPTION TAKE THRUST
PARAMETERS AND
RECOMPUTE TIG

HOLD

SNAP

FLASH VERB-NOUN TO
REQUEST RESPONSE AND
DISPLAY TIG
V06 N33
R1-TIG-HRS
R2-TIG-MIN
R3-TIG-SEC

TIG-TIME OF IGNI-
TION (GET).
IN HRS, MIN, SEC TO
NEAREST .01 SEC

MONITOR DSKY:
OBSERVE VERB-NOUN
FLASH TO REQUEST
RESPONSE AND DISPLAY
OF TIG

RECORD DATA

WAIT FOR KEYBOARD
ENTRY

KEY IN PROCEED

```
        TERMINATE FLASH UPON
        RECEIPT OF PROCEED
        ------------------
                  .P
                  .R
                  .O
                  .C
                  .E
                  .E
                  .D
                  .
                  .
                  .
        ------------------
        IS REFSMMAT FLAG
        SET?
        ------------------
                  .Y      .N
                  .       .
                  .       .
        ----------------  .
        COMPUTE IMU GIMBAL.
        ANGLES AT IGNIT-  .
        ION FOR THRUST    .
        ALONG CSM +X AXIS .
        ----------------  .
                  .       .
                  .       .
                  .       .
                  . ----------
                  . SET MGA
                  . DISPLAY IN
++                . R3(BELOW)
+20               . = -00002
++                . ----------
                  .       .
                  .       .
                  .       .
        ------------------                     ------------------
HOLD  . FLASH VERB-NOUN TO                   . MONITOR DSKY:
....... REQUEST RESPONSE AND ................. OBSERVE VERB-NOUN
MON   . DISPLAY MANEUVER                     . FLASH TO REQUEST
        DATA:                                  RESPONSE AND DISPLAY
          V16 N45                              OF TFI AND MGA
          R1-MARK CTRS                         ------------------
          R2-TFI                                        .
          R3-MGA                                        .
                                                        .
++      MARK CTRS-NOT MEAN-                             .
+20     INGFUL TO THIS                                  .
++      PROGRAM.                                        .
                                                        .
                                                        .
                                                        .
                                                        .
                                                        .
                                                        .
                                                        .
```

TFI - TIME FROM TIG.
IN MIN AND SEC TO
NEAREST SEC.
MAXIMUM READING IS
59B59 (- BEFORE +
AFTER TIG)

MGA:MIDDLE GIMBAL
ANGLE AT TIG IF
+X CSM AXIS IS
ALIGNED WITH INITIAL
THRUST DIRECTION.
SIGN IS ALWAYS +
++ EXCEPT WHEN THE IMU
+20 IS NOT ALIGNED THE
++ VALUE IS -00002. IN
DEGREES TO NEAREST
.01 DEGREE

WAIT FOR KEYBOARD ENTRY

KEY IN PROCEED

TERMINATE FLASH UPON
RECEIPT OF PROCEED

PROCEED

RESET EXTERNAL DELTA
V FLAG

DO ROUTINE R00

DO ROUTINE R00
SELECT PROGRAM
CORRESPONDING TO
PROPULSION SYSTEM
SELECTED PREVIOUSLY
(SPS-P40, RCS-P41).

EXIT
P37

EXIT
P37

#790

CHANGE CONTROL NOTES:

LOGIC REV 19	PCR NASA 108	
	PCR NASA 109	#800
LOGIC REV 20	PCR MIT 66	

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